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CONFERENCE ON CERTIFICATION OF AIRCRAFT FOR LIGHTNING AND ATMOS--ETC(U)
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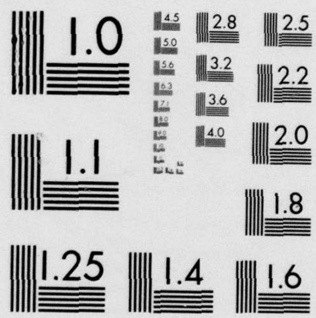

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CONFERENCE ON CERTIFICATION OF AIRCRAFT FOR LIGHTNING AND ATMOSPHERIC ELECTRICITY HAZARDS

(CONFÉRENCE SUR LA CERTIFICATION DES AÉRONEFS CONTRE LES DANGERS DU FOUDROIEMENT ET DE L'ÉLECTRICITÉ ATMOSPHÉRIQUE)

held

ONERA - CHATILLON (France)
14-21 septembre 1978

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20. Abstract

This conference was organized to disseminate in a tutorial fashion to the NATO Air Electrical Working Party and their technical advisors the information related to the recent progress of atmospheric electricity hazards and lightning protection studies.

The conference program covers the following topics: Theory of Lightning; Basis of Specification; Lightning Attachment and Swept Stroke Testing; Fuel Vapor Ignition and Direct Effects Testing; Indirect Effects Testing and Lightning Protection Methodology; Static Electricity.

FORM 1473

EOARD-TR-79-6 DATE: 29 November 1979

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This technical report has been reviewed and is approved for publication.

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PROCEEDINGS

OFFICE NATIONAL D'ÉTUDES ET DE RECHERCHES AÉROSPATIALES
29, Avenue de la Division Leclerc, 92320 CHATILLON (France)

1 – BACKGROUND OF THE CONFERENCE

The NATO Air Electrical Working Panel (AEWP) was convened to meet September 14-21, 1978 in Paris, France, in order to consider international standards for the certification of aircraft for atmospheric electricity hazards and lightning protection. Because of the extent and the complexity of the relevant technology developed within the past 5 years, it was considered essential to disseminate this information in a tutorial fashion to the AEWP members and their technical advisors. This was the purpose of the above conference held at l'Office National d'Etudes et de Recherches Aérospatiales (ONERA), Châtillon, France, from September 11-13, 1978.

2 – ACKNOWLEDGEMENTS

It is a pleasure to acknowledge the support of the Office National d'Etudes et de Recherches Aérospatiales (ONERA), and the sponsorship, of the European Office of Aerospace Research and Development (EOARD), Air Force Systems Command, United States Air Force, and the Naval Air Test Center, United States Navy under Grant AFOSR-78-3653. The participants and the organizers wish to express their appreciation for the smooth functioning of the Conference and the preparation of the Proceedings to M. Joseph Taillet and to the many ONERA participating personnel, including MM. Claude Sevestre and Alfred Fa'arman.

3 – ORGANIZATION

The planning and arrangements were entrusted to Dr Joseph Taillet and Lt Col. Todd Newell, Conference Administrators and a steering committee composed of :

Conference Steering Committee

Mr. Gary Dubro (US)
Dr. Joseph Taillet (FR)
Dr. Philip Little (UK)
Lt. Col. Hans Melcher (GE)
Mr. Charles Seth (US)
Mr. Robert Evans (UK)
Mr. Philippe Lecat (FR)
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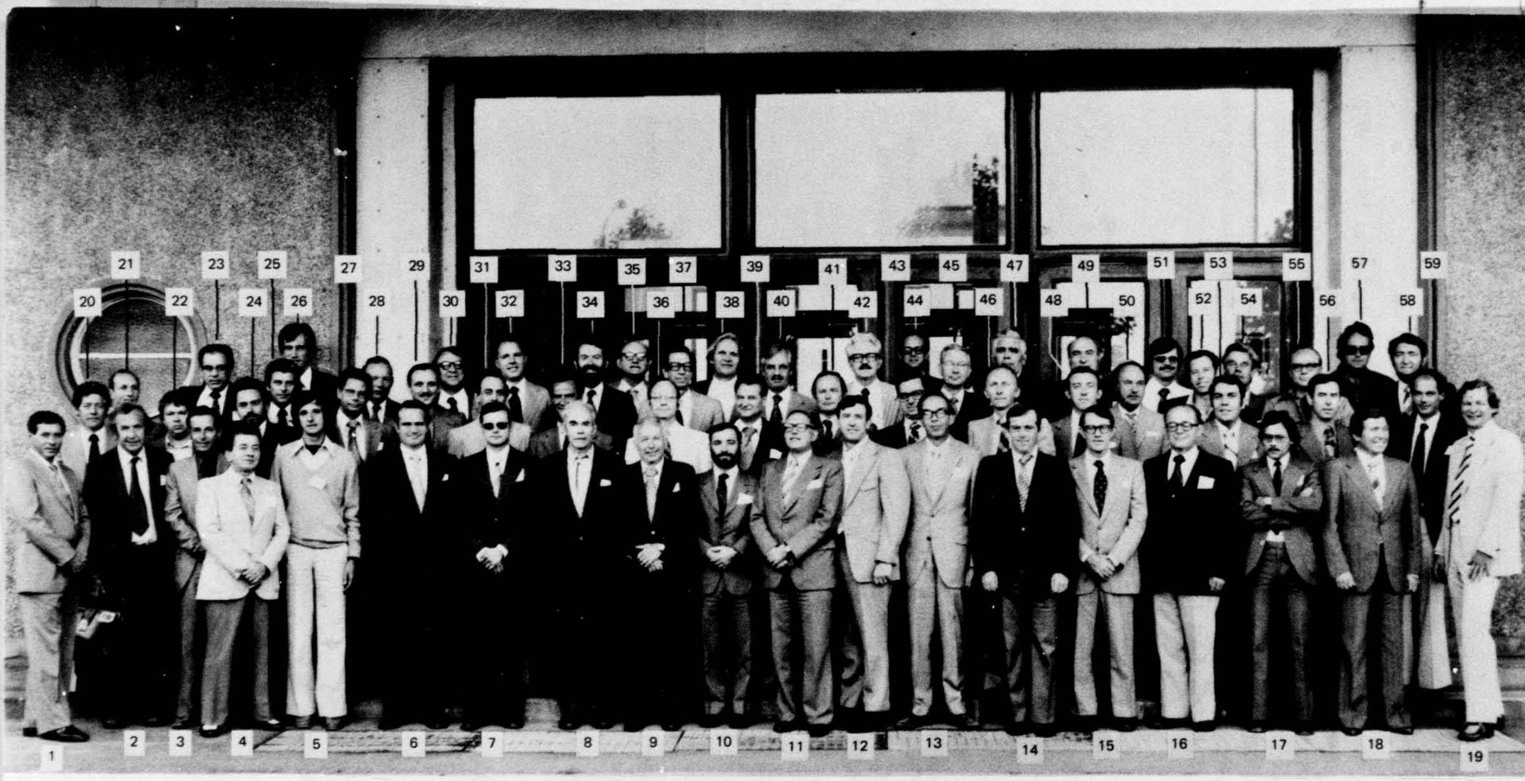

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NATO UNCLASSIFIED

CLOSED CONFERENCE ON

CERTIFICATION OF AIRCRAFT FOR LIGHTNING AND ATMOSPHERIC ELECTRICITY HAZARDS

11-13 September 1978

Held at : Office National d'Etudes et de Recherches Aérospatiales (ONERA)

29, avenue de la Division Leclerc
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4 – WELCOMING REMARKS

The welcoming remarks for the host, ONERA, were given by M. l'Ingénieur Général Pierre Contensou, Director General of ONERA.

*

It is an honor for l'Office National d'Etudes et de Recherches Aérospatiales (ONERA) to serve as the host for this Conference on the electrostatic protection of aircraft. The choice that you have made, in organizing this scientific meeting in France, is in effect a demonstration of the interest shown by the international technological community for the recent programs of our scientists, in a field where the USA and the UK have paved the way.

The future development of generalized automatic control and of "fly by wire" necessitate a precise knowledge of the interaction of atmospheric electricity, in all its forms, with the critical flight control circuits ; the operation and the survivability of these flight control circuits must be guaranteed by the most efficient measurements, controlled by rigourous ground test procedures. It is for the purpose of preparing the future selection of standards by the NATO Air Electrical Working Panel (AEWP) that the experts of the various nations of NATO have been convened at this Conference, in order to present the physical and technological "state of the art", which should orient that selection.

I trust, therefore, that this meeting will be most successful, and that its conclusions will serve as a basis for future implementation.

At a recent conference on earthquake phenomena, the chance appearance of an unexpected tremor caused the participants to pass from the domain of theory to that of practice. The merit of the experimenters present here, and their sense of practical application, explains why ONERA has not considered it necessary, in the course of this meeting, to further inspire them by triggering a lightning stroke on this building in Châtillon.

Welcome to each one of you, and best wishes for a very successful conference.

*

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PROGRAM

Welcoming address by *Pierre Contensou,* Director General of ONERA.

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LIGHTNING PHENOMENA; THEORY AND BACKGROUND

J. E. Nanevicz
Program Manager
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333 Ravenswood Avenue
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Electrical processes associated with thunderclouds are discussed from the point of view of the aircraft designer concerned with questions such as physical damage to the aircraft, and the electromagnetic effects of the lightning flash on avionic systems.

I INTRODUCTION

A. General

An active thundercloud has associated with it a variety of electrical processes of great concern to the aircraft designer and operator. Currents as high as 100,000 Amperes flow in the lightning stroke channels transferring charge of up to 200 Coulomb. High static electric fields exist in the vicinity of the cloud, and can induce substantial streamer currents from the extremeties of a vehicle flying nearby. Processes associated with the formation and propagation of a stroke generate high-level electromagnetic-transient signals in the vicinity of the storm cell.

B. Motivation for the Study of Lightning Phenomena

Since aircraft must operate in the vicinity of thunderstorms, they must be designed and tested to survive the electrical activity associated with the storms. This means that the structure of the aircraft must be such that it can pass the maximum expected stroke current without damage. Conducting paths, such as antenna lead-ins or fuel filler caps, which might conduct lightning currents to the interior of the aircraft must be equipped with devices capable of bypassing the expected stroke current to the exterior of the skin without damage. Underdesign of the structure or of lightning protection devices renders the aircraft vulnerable to risk of damage. Overdesign, on the other hand, generally implies an unnecessary weight and cost penalty, and is also unacceptable. Accordingly, work was undertaken to gather data on lightning current waveforms and charge transfer to devise a "standard" stroke that duplicates in the laboratory the damage observed on actual strikes to aircraft.

For the past several decades, aircraft have generally been fabricated using all-metal skins, with the specific materials and dimensions determining by structural requirements. The metal skin material, thickness, and fabrication techniques were usually such that an aircraft could pass lightning stroke currents with relatively minor damage--usually pitting and puddling of the skin at the stroke attachment points. (Occasionally, it was necessary to provide extra skin thickness in critical regions such as fuel cells.) In addition, metal skins shielded the interior electronic systems from interference signals generated by precipitation static, nearby lightning, and direct lightning strikes. Aircraft are currently being fabricated with many of their complex, non-load-bearing surfaces made of fiberglass. In the future, it is planned that for reasons of strength, weight, and cost, large sections of aircraft skin and structure will be made of composite materials. These nonmetallic materials are not suitable for conducting lightning stroke currents and do not provide the RFI shielding that was formerly obtained with metal structures and skin.

Programs are currently under way to devise techniques, such as coatings, to improve the electrical and shielding properties of composite materials. The implementation of these special processes may impose significant penalties--weight, initial cost, and increased maintenance cost. Accordingly, an aircraft designer requires accurate information about the physical processes for which he is endeavoring to provide protection, so that he can avoid overdesigning or underdesigning.

Electronic systems used on early aircraft used high-level analog circuitry that was tolerant of occasional noise pulses of considerable amplitude. In addition, the aerodynamic design of the aircraft was such that functioning of the avionic systems was not essential to achieve stable flight. Present solid-state digital electronic systems operate at lower signal levels and suffer component damage and upset at much lower levels of noise pulses. Ultimately, it is planned that all avionic systems will be completely digital and will operate at even lower signal levels than are currently used. Experience with digital systems indicates that they are often susceptible to catastrophic upset as the result of a single noise impulse at a level comparable to the system operating level. Furthermore, it is likely that the next generations of aircraft will rely on stability-augmentation or fly-by-wire avionic systems to such an extent that the airplane cannot be flown without them. For these reasons, increasing attention must be paid to the signals that the lightning activity induces in wiring and avionic systems on the interior of the aircraft.

Experiment and analysis indicates that the high-frequency currents in the lightning stroke are responsible for the signals penetrating through apertures in the skin to the interior wiring of the aircraft avionic systems. Accordingly current activity in lightning characterization includes substantial emphasis on defining the nature of the H-F components, and in devising ways of duplicating them in ground simulation tests.

II THE THUNDERCLOUD

A. Physical Processes and Cloud Form

The classification of a storm as a thundercloud or thunderstorm requires that thunder be heard which, in turn, implies the presence of lightning.[1*] These storms are composed of strongly convective cumulonimbus clouds generally accompanied by strong wind gusts and rain, or sometimes hail or snow. Thunderclouds usually develop as the warm, moist air near the earth rises and replaces the denser air aloft. As a consequence of this overturn, the condensation of atmospheric water vapor occurs forming a visible cloud of water droplets. The heat associated with the phase changes of water speeds the overturn: release of the heat of vaporization by condensing water vapor enhances the updrafts, while cooling, caused by evaporation of condensed water, can help drive the downdrafts which replace some of the ascending subcloud air. These air motions are illustrated in Figure 1. Since the interiors of thunderclouds are highly turbulent and hazardous, good information is not available on conditions within their boundaries.

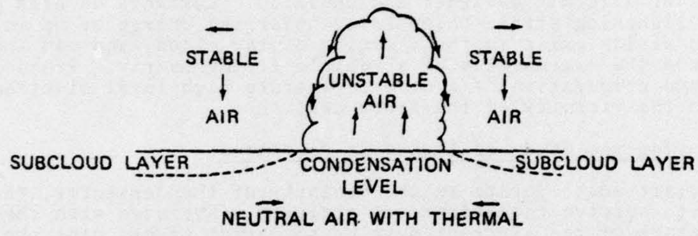

FIGURE 1 THE MOTIONS OF AIR IN AND AROUND A CUMULUS GROWING INTO STABLE AIR (AFTER VONNEGUT AND MOORE)

Active cumuli continue to grow in volume and height until they encounter a thermally stable layer in the atmosphere which serves to limit further vertical development. Frequently, strong horizontal winds exist at cloud top altitudes; these blow the ice crystal cloud downwind forming an anvil cloud of the sort illustrated in Figure 2 which was photographed during recent flight test studies of thunderstorm static electric fields in Florida.[2]

B. Electrification Processes, Charge Distribution, and Static Fields

Cloud physicists are not in agreement regarding the mechanisms responsible for thunderstorm electrification. It has generally been assumed that negative charge is selectively separated and transported downward by falling precipitation particles. Mechanisms proposed for producing the charge separation include induction charging resulting from particle collisions in an existing electric field, to charge separation occurring as the result of ice crystal splintering.[1] The study of particle electrification is an active area of cloud physics.

An early electrostatic model for the distribution of charge in a South African thundercloud suggested by D. J. Malan is shown in Figure 3.[3] It was constructed using ground measurements of electric field intensity in the vicinity of thunderclouds. The static electric field structure resulting from this distribution of charges is shown in Figure 4.[2] The length of the small arrows in the figure is proportional to the log magnitude of the potential gradient at the origin of each arrow; the arrow direction shows the direction of the maximum change in positive potential. (This is the atmospheric electrician's convention in which the arrows point toward positive charge.)

Recent flight test measurements by SRI indicate that the fields in the vicinity of Florida thunderstorms are not given by the model of Figures 3 and 4. Higher charge magnitudes and a substantially more complex electrostatic model such as that suggested in Figure 5 are required to account for the observed fields in the vicinity of the cell and in the region of the anvil.[2]

Although electrostatic charge models are appealing for a variety of reasons, (they are simple in concept, easy to use, and permit a particularly simple linkage of external field observations to internal charge magnitudes) researchers have pointed out that the use of such models for thunderstorm charge distributions gives erroneous results if (as is generally believed) the electrical conductivity of the cloud and the surrounding atmosphere are functions of position.[4-6] Accordingly, substantial work is currently underway to develop more accurate models of the charge distribution within thunderstorm clouds. Such models are needed to more completely define the static fields about thunderclouds to permit the assessment of the likelihood of triggering lightning by the introduction of a large vehicle such as an aircraft or rocket.

*References are listed at the end of the paper.

FIGURE 2 MATURE THUNDERSTORM WITH FULLY FORMED ANVIL (21 JULY 1975, CLOUD D)

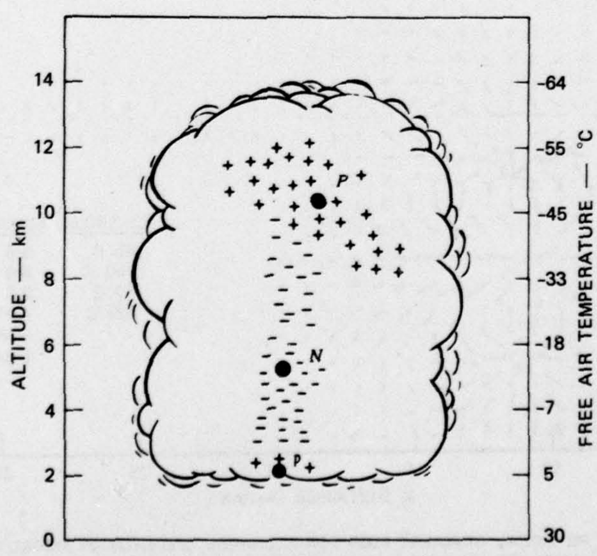

FIGURE 3 PROBABLE DISTRIBUTION OF THE THUNDERCLOUD CHARGES, ACCORDING TO MALAN. Solid black circles indicate locations of effective point charges, typically P = +40 coul at 10 km, N = -40 coul at 5 km and p = +10 coul at 2 km to give observed electric field intensity in the vicinity of the thundercloud.

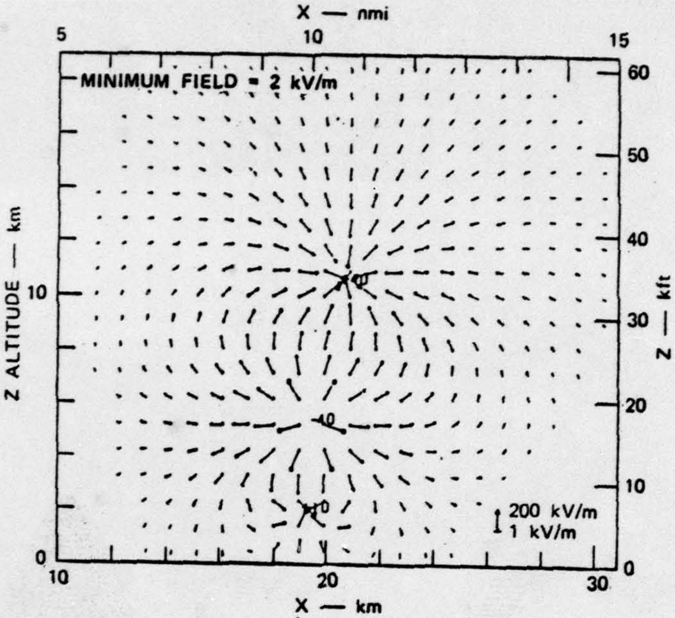

FIGURE 4 POTENTIAL-GRADIENT VECTOR PLOTS FOR MALAN MODEL CHARGE DISTRIBUTION

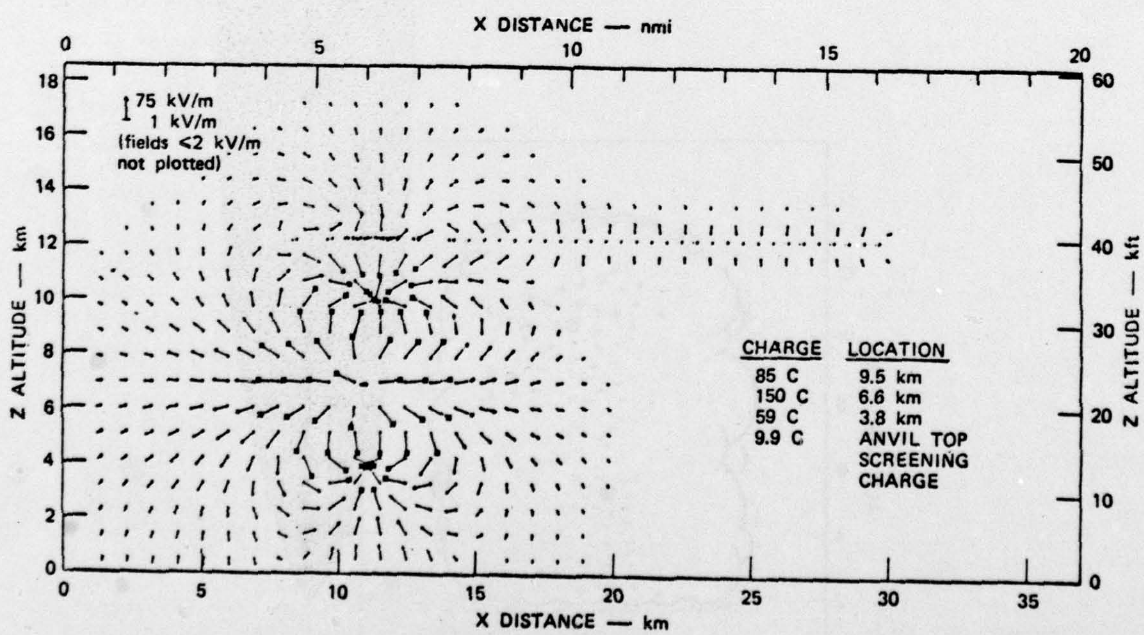

FIGURE 5 X-Z PLANE POTENTIAL GRADIENT PLOT FOR A CHARGE DISTRIBUTION MODEL INVESTIGATED FOR COMPARISON WITH SRI LEARJET FLIGHT TEST DATA

III LIGHTNING DISCHARGE PROCESSES

Lightning is a transient high-current electric discharge usually several kilometers in extent. Discharges can occur from the cloud to ground or they may be entirely within the same cloud--intracloud flashes. As shown for the case of a cloud-to-ground discharge in Figure 6, a lightning flash involves a large number of processes and should be expected to generate a complex electromagnetic signature.[3]

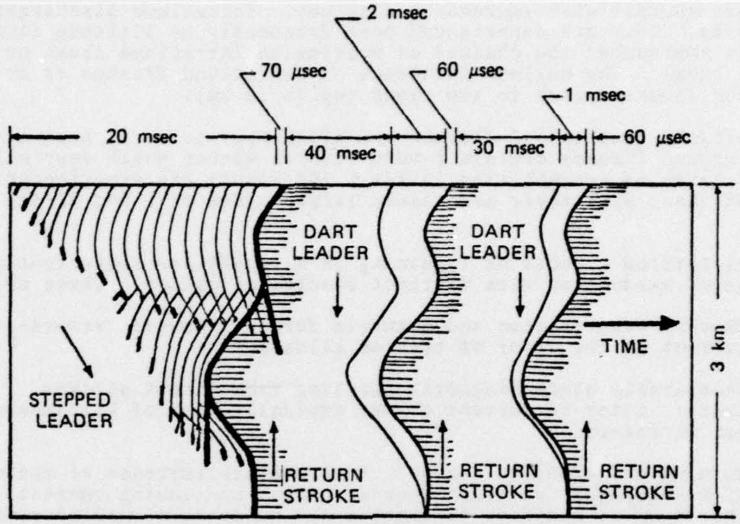

FIGURE 6 THE LUMINOUS FEATURES OF A LIGHTNING FLASH AS WOULD BE RECORDED BY A CAMERA WITH FIXED LENS AND MOVING FILM. INCREASING TIME IS TO THE RIGHT.

A total lightning discharge (duration $\approx$ 0.5 sec) is called a flash and is composed of three or four component discharges (luminous for $\approx$ 100 μs) called strokes separated by about 40 ms. Each stroke consists of a weakly-luminous leader which propagates to the ground followed by a very luminous return stroke which propagates from the ground to cloud.

The first cloud-to-ground predischarge in a flash is called the stepped leader. The stepped leader appears to move downward in luminous steps of roughly 50-m length with a pause time between steps of about 50 μs. During the pause time the stepped-leader channel is not luminous enough to be recorded on photographic film using standard streak-camera techniques. Each leader step becomes bright and observable in a time less than a microsecond. In Figure 6 the 50-m steps appear as darkened tips on the faintly luminous channel extending upward into the cloud. Typically the average velocity of the stepped leader during its trip to the ground is 1.5×10^5 m/s. Peak currents in a leader steps are of the order of 10^3 A.

When the stepped leader has lowered a charged column of high negative potential to near the ground, the resulting high electric field at the ground is sufficient to cause upward-moving discharges to be initiated from the ground or from objects on the ground toward the leader tip. When one of these discharges contacts the leader, the bottom of the leader is effectively connected to ground potential while the remainder of the leader is at negative potential and is negatively charged. The situation is somewhat similar to a transmission line charged to a constant potential with a short circuit applied at its end. The leader channel acts like a transmission line (nonlinear) supporting a very luminous return stroke.[7-9] The return-stroke wavefront, an ionizing wavefront of high electric field intensity, carries ground potential up the path produced previously by the stepped leader. The return-stroke wavefront propagates at a velocity of typically one-third to one-tenth the speed of light, making the round trip between cloud base and ground in a time of the order of 70 μs. The region between the return-stroke wavefront and ground is traversed by large currents. The net negative charge deposited on the leader channel is effectively lowered to earth through the highly conducting channel beneath the return-stroke wavefront.

Once the stroke current has ceased to flow, the lightning flash may end. However, if additional charge is made available to the top of the channel, the flash may contain additional strokes (a multiple-stroke flash). If additional charge is made available to the decaying return-stroke channel in a time less than about 100 ms, a continuous or <u>dart leader</u> will traverse that return-stroke channel, increasing its degree of ionization, depositing charge along the channel, and carrying the cloud potential earthward once more. The dart leader thus establishes the conditions for the second return stroke. The dart leader appears to be a luminous section of channel about 50-m in length which travels

smoothly earthward at about 2×10^6 m/s, an order of magnitude faster than the average velocity of the stepped leader.

IV LIGHTNING STROKE PARAMETERS

A. General

Both intracloud and ground flashes are probably initiated within the cloud in restricted areas of very high electric field. Typically, these areas are concentrated around an altitude of some 3 km with the cloud base being at 1 km. It follows that the probability of an aircraft intercepting a flash to ground is almost uniform from 0 to 3 km and then drops off sharply with increasing altitude. Intracloud discharges begin to be encountered at 1 km. They are experienced more frequently as altitude increases, and as the 3-km level is approached the chances of meeting an intracloud flash or a discharge to ground are about equal. The maximum incidence of intracloud flashes is at about 6 km, and few intracloud flashes reach to the cloud top ($\sim$ 12 km).

Electrically, intracloud flashes and discharges to earth have one major difference. Cloud-to-ground flashes contain return strokes within which very high peak currents ($i \sim 100$ kA) and rates of current rise ($di/dt \sim 100$ kA/μs) are experienced. There are no true return strokes, with their associated large values of i and di/dt, in intracloud discharges.[10]

The deleterious effects of lightning on aircraft are conveniently separated into four categories associated with distinct electrical causes. These are:

(1) Thermal vaporization and magnetic forces. Cause: return-stroke current of the order of tens of kiloamperes.

(2) Undesirable electromagnetic coupling from direct strokes. Cause: rates of current change typically tens of kiloamperes per microsecond.

(3) Burning and erosion. Cause: intermediate currents of the order of kiloamperes for milliseconds. Also, continuing currents of the order of hundreds of amperes for hundreds of milliseconds.

(4) Electromagnetic coupling from flashes that are "near misses."

Both intracloud discharges and flashes to earth are almost equally potent as regards Effect 3. For Effect 4, over most frequencies there is little difference between the two types of discharge. However, Effects 1 and 2 are dominantly produced by return strokes and therefore by flashes to earth. Tests geared to the severity of flashes to ground will therefore adequately cover intracloud discharges.

B. Statistics of Lightning Parameters

Although many parameters are required to define all of the processes involved, many are of little importance in causing hazards. Some other parameters are of much greater importance, and for them, our knowledge must be constantly updated. Two important parameters for the return stroke in the flash to earth are the peak current, i_p, and the rate of current rise, di/dt (most appropriately defined as the average value of di/dt over the rise (front) time T_F from $i = 0$ to $i = i_p$). Another important return-stroke parameter is the half-value time occupied in decaying from the peak i_p to $i = 0.5\, i_p$. The statistics of return-stroke parameters are represented in Figure 7 for the usual flash transporting negative charge to ground.[10] The representation is conveniently formulated in terms of the log-normal distribution; this distribution is closely obeyed by many parameters. Figure 7 terminates (as do Figures 8 and 9) at the 2% point; the distribution is easily extrapolated to more extreme values, but the greater the extrapolation the greater the uncertainty.

The main surge of return-stroke current is usually followed by an "intermediate" current of a few kiloamperes lasting for a few milliseconds. Although measurements of intermediate currents have been made, and characteristics of the currents have been deduced from observations of atmospherics, no statistical information on intermediate currents is readily available. This is unfortunate, since it is believed that intermediate currents are the most likely type of current to produce metallic puncture when--as is common with aircraft--the point of flash attachment is being swept by the windstream.

Most discharges include a phase of continuing current. Intracloud flashes consist predominantely of continuing current; the superimposed K recoil surges represent only minor perturbations. Even for the discharge to ground, continuing currents rather than return-stroke surges produce most of the charge transfer. Statistics for continuing currents are shown in Figure 8.

Many lightning parameters are derived, rather than basic. Two of these, often quoted by engineers, are the total charge transfer Q per flash ($\int i\, dt$), and the total action integral, I, per flash ($\int i^2\, dt$). Distributions for these parameters are given in Figure 9. Note that the distribution for total charge transfer is rather more extreme than that for the charge passing in continuing currents, although most charge in a typical flash flows during a continuing current. Some flashes, however, contain no continuing current phase at all, while some may include two or more such phases. These facts account for most of the difference between the two charge distributions.

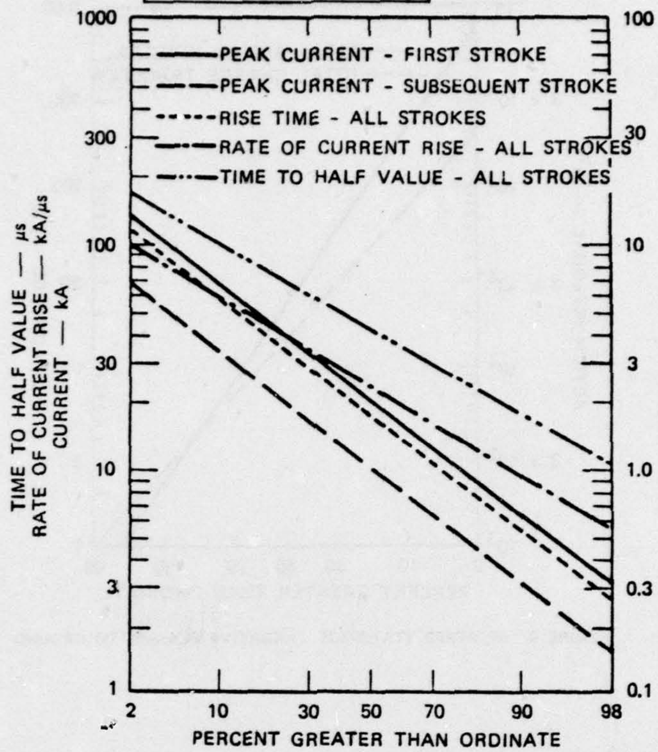

FIGURE 7 STATISTICS FOR RETURN-STROKE PARAMETERS — NEGATIVE STROKES

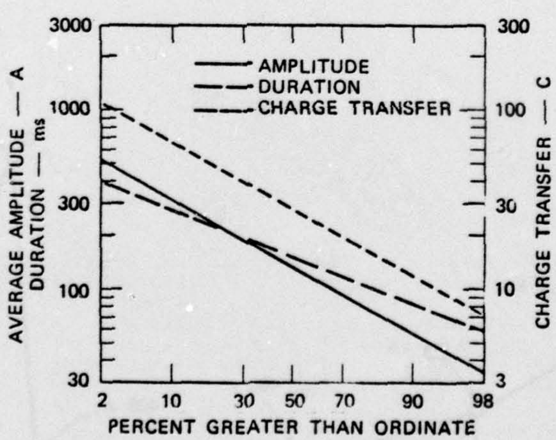

FIGURE 8 STATISTICS FOR CONTINUING CURRENTS - NEGATIVE FLASHES TO GROUND

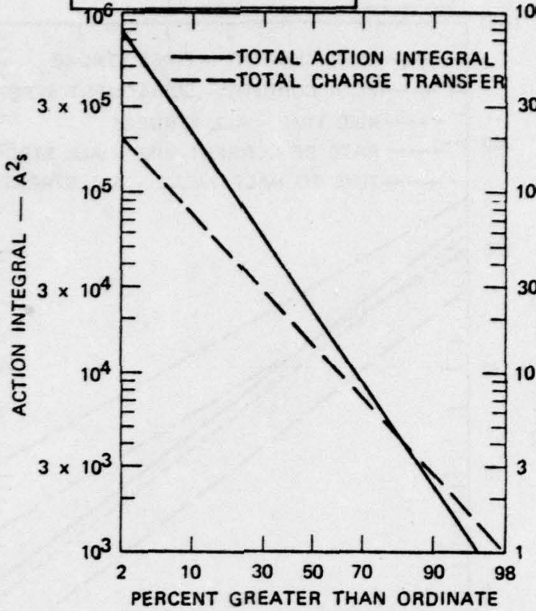

FIGURE 9 FURTHER STATISTICS - NEGATIVE FLASHES TO GROUND

C. Lightning Model

In conducting lightning testing, it is essential to have a model representing a typical flash. Figure 10 shows a model representation of a very severe lightning flash devised in connection with work on Space Shuttle.[11] The model parameters are selected to be at comparable statistical levels. The representation is intended only as a guide and has therefore been much simplified in both physical and analytical respects. For instance, straight lines connect the key points (A, B, C, etc.) in the current time history. However, oversimplification is usually preferable to unjustifiable overcomplication.[12]

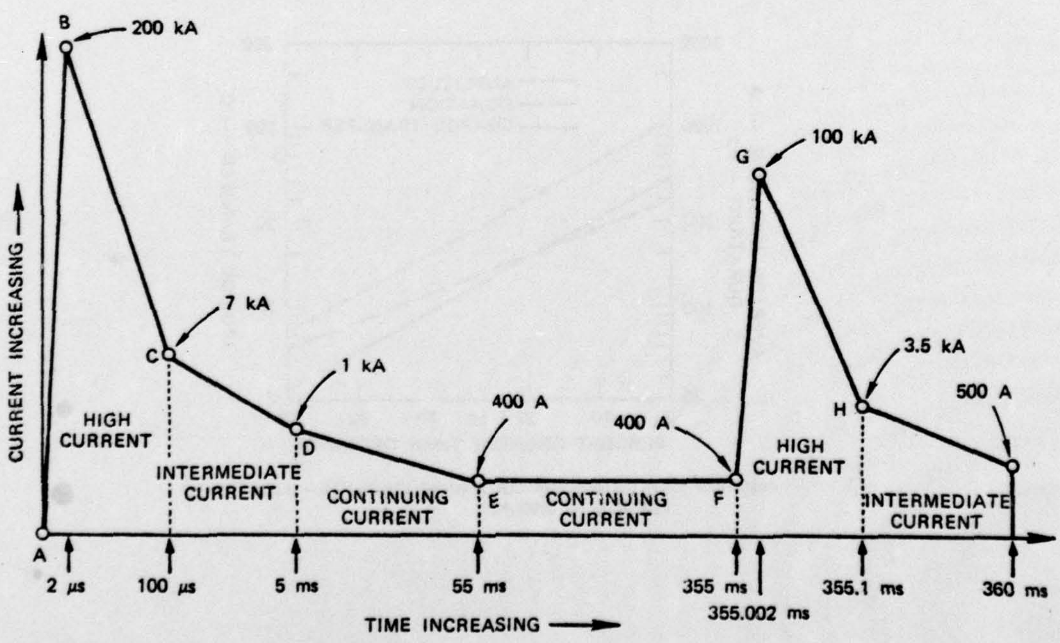

FIGURE 10 SEVERE LIGHTNING MODEL FOR SPACE-SHUTTLE WORK

V ELECTROMAGNETIC SIGNALS ~~FROM LIGHTNING~~

From Figures 6 and 10, it is evident that a lightning flash is made up of a number of different components, and should generate a complex electromagnetic signature. The structure of the electromagnetic radiation from lightning varies with frequency and time as schematically illustrated in Figure 11.[13] At very low frequencies, VLF (3 to 30 kHz),

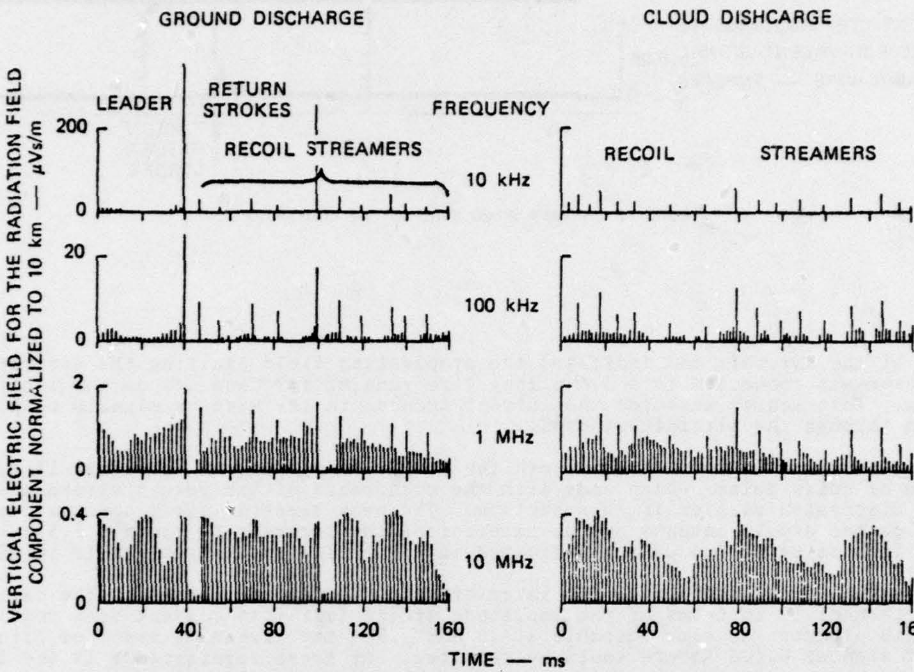

FIGURE 11 STRUCTURE OF THE FIELDS RADIATED BY LIGHTNING AS A FUNCTION OF TIME AND FREQUENCY

the pulses are discrete and are generated principally by the return stroke and/or recoil streamers (K-charges). As the frequency increases, the number of pulses per flash also increases, with a maximum of about 10^4 per discharge for frequencies between 30 to 300 MHz (VHF); the disturbance accompanying the flash is then quasi-continuous. These pulses appear to be associated with the initial leader, including its steps, and also with the electrical-breakdown processes accompanying probing leaders moving within the cloud. These probing leaders can occur, for a flash to earth, between return strokes or after the final stroke; for an intracloud discharge their presence is possible at almost any stage of the discharge. It is interesting to note that the signals at HF and VHF associated with return strokes and K changes are not strong, and are indeed partly "quenched" following the occurrence of return strokes and K changes. It is believed that this quenching is due to a temporary absence of probing leaders. As frequency is further increased beyond the VHF range, there is a sharp decrease in the number of pulses until at centimetric wavelengths (∿ GHz) the pulses are again well separated and associated with the macroscopic features of the return strokes.

It cannot be over emphasized that a lightning flash involves a multiplicity of sparks and consequently a protracted and complicated generation of radio signals. It is a common misconception that the lightning discharge occurs as one single large spark, and that therefore all radio emissions are produced almost simultaneously as in the case of the nuclear electromagnetic pulse (EMP). In reality the time histories of the lightning emissions and of the EMP are quite different, and comparisons of equipment responses to the two types of signal should entail intelligent recognition of this fact. For example, at HF the EMP would generate one very large pulse while a typical lightning flash might create ten thousand small pulses. Designs of equipment can be conceived that could survive the single large pulses but would fail under the repetition of the small pulses.

Finally, in applying the results of ground measurements to the analysis of the aircraft lightning environment it is important to note that the subsidiary sparks providing most of the radiation at HF and VHF are usually located within the thundercloud; consequently, they are seldom very close to ground equipment, but may be close to an airplane. For this reason, flight test programs have been undertaken recently to better define the airborne environment.[14-16]

A sample of data from one such program is shown in Figure 12.[16] The two records shown are the outputs from a pair of spectrum analyzer receiver channels with center frequencies of 10 MHz. One channel was connected to an electric dipole antenna on the

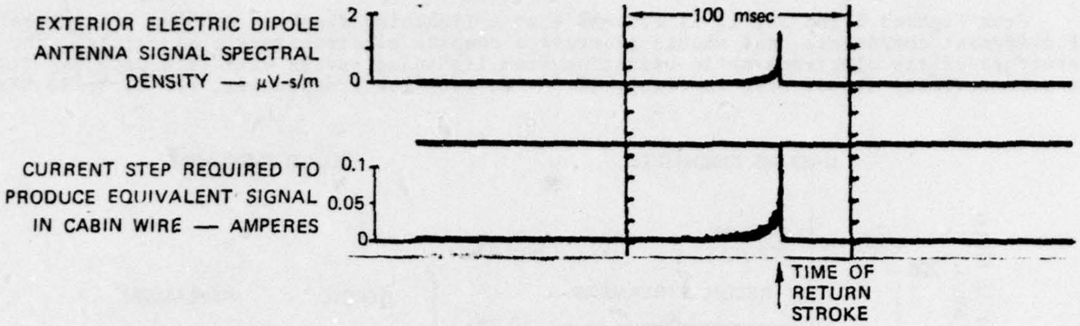

FIGURE 12 - HIGHER SPEED STRIPOUT OF LIGHTNING

exterior of the aircraft and indicated the propagating field exciting the aircraft. The other sensor was connected to a 3.6-m long wire running fore and aft on the interior of the cabin. This sensor measured the current induced in the wire by signals coupling into the cabin through the aircraft windows.

It is interesting to note that both the 10 MHz signals shown in Figure 12 exhibit a crescendo of noise pulses which ends with the occurrence of the return stroke in agreement with the discussion earlier in this section. The peak electric-field spectral density measured on the dipole antenna on the exterior of the aircraft is roughly 1.5 μV-s/m compared to a value of 0.4 μV-s/m indicated in Figure 11 for a discharge 10 km distant.

The signal level induced on the interior wire is also of interest. The calibration for this channel is in terms of the amplitude of the fast-rise current step required on the wire to produce the same response at 10 MHz. For the lightning event of Figure 12, a current step of 0.125 Ampere would be required. Of great significance is the fact that a direct strike to the Learjet test aircraft produced a reaction equivalent to a 1-Ampere current step in the cabin wire. Nearby lightning events produced 10 MHz noise signals equivalent to 0.5 Amp current steps.

The flight test results are highly significant in several respects. First, they confirm the discussion earlier in this section. Secondly, they point out that substantial H-F signals can be generated in aircraft wiring by nearby lightning. Since the equivalent pulse levels are of almost the same amplitude, and since nearby lightning events are much more numerous than direct strikes, nearby strokes should not be ignored in lightning interference analyses. Third, they indicate that the duration of the noise signals induced in interior wiring can be substantial--over 20 ms in Figure 12.

VI LIGHTNING TRIGGERING

Triggered lightning is defined as lightning that is caused by some human modification of the natural atmospheric environment.

According to E. T. Pierce, almost all instances of lightning triggered by man, either by design or by accident, involve the introduction of a long electrical conductor into a thundery environment where the general electric field is some 10 kilovolts per meter (kV/m). If the potential discontinuity between the tip of the conductor and the ambient atmosphere becomes about a million volts (10^6 V) a leaderstreamer is initiated from the conductor and triggered lightning can occur.[17]

Although aircraft are often struck by lightning it is usually impossible to determine whether the flash was initiated by the presence of the aircraft or not. However, there are at least two published accounts of several instances in which the lightning was almost certainly triggered by aircraft. In most of these cases the flashes occurred while aircraft were penetrating dissipating thunderstorms which had ceased to produce lightning; the only discharges that took place during these penetrations were to the aircraft. It is also understood that during the development of the Boeing 747, test flights were made in which the aircraft was flown in and out of clouds that were strongly electrified but not actually generating lightning. The 747 was consistently struck soon after entering the clouds; no other lightning, except that to the 747 occurred.

If we accept that a certain voltage discontinuity is necessary in order to initiate a leader-streamer it follows that the bigger an aircraft (greater ℓ) the more likely it is to trigger lightning. There are indeed indications that, under comparable environmental conditions, the large aircraft now coming into widespread service are struck more frequently than their smaller predecessors. However, the available statistics are limited in extent and need careful interpretation. There is also some evidence that high-speed aircraft experience more lightning strikes than their slower fore-runners.

It is interesting to examine a lightning strike to the NASA Learjet used for thunderstorm static field studies in 1976 in light of lightning triggering criteria.[14] On run 8 of the 10 August 1976 test when the aircraft was passing between the main set of cells and a single cell developing to the right, it was struck by lightning. The stroke attachment points on the aircraft were the nose radome, which was damaged and had to be replaced, and the aft end of the right wing tip fuel tank, which was burned and pitted.

A record of the static electric field readings recorded during this run is shown in Figure 13. Also shown in the figure is a rough sketch of the weather radar display showing a storm cell on each side of the aircraft.

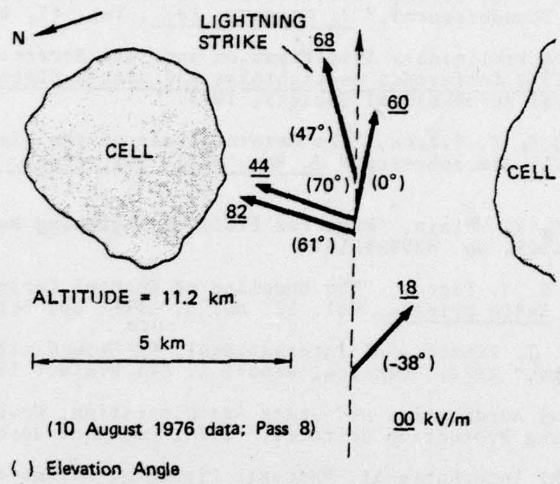

FIGURE 13 ELECTRIC FIELD VECTORS OBSERVED ON A PASS DURING WHICH THE LEARJET TEST AIRCRAFT WAS STRUCK BY LIGHTNING

The magnitude and direction of the ambient static field prior to the strike ($\approx$ 70 kV/m directed along a line 15° to the left of the aircraft roll axis and elevated 47° above the horizontal) should be noted in the light of E. T. Pierce's criterion for necessary conditions for a conducting body to trigger a stroke. The ambient field was well in excess of 10 kV/m. The overall length of the Learjet is 13.2 meters so that the aircraft shorted out a total potential of 924 kV. Thus it is possible that the rapid introduction of the aircraft into the region of high electric field triggered the stroke.

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12. E. T. Pierce, SRI International, "Natural Lightning Parameters and Their Simulation in Laboratory Tests," 1976, Scientific Note E, SRI Project 3062.

13. N. Cianos, G. H. Oetzel, and E. T. Pierce, "Structure of Lightning Noise--Especially Above HF," Lightning and Static Electricity Papers at Las Vegas, Nevada, Air Force Avionics Laboratory, Wright Patterson AFB, Ohio, 1972.

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15. Jerome T. Dijak, Air Force Flight Dynamics Laboratory, "Airborne Measurements of Transient Electric Fields and Induced Transients in Aircraft Due to Close Lightning," 1977, Project 43630135.

16. J. E. Nanevicz, R. C. Adamo, E. F. Vance, and R. T. Bly, Jr., "Florida Lightning Electromagnetic-Field Measurements Using an Instrumented NASA Jet Aircraft," Federal Aviation Administration/Georgia Institute of Technology Workshop on Grounding and Lightning Protection, May 1978.

17. Edward T. Pierce, "Triggered Lightning and Some Unsuspected Lightning Hazards," American Association for the Advancement of Science 138th Annual Meeting, Philadelphia, December 1971.

QUESTIONS and ANSWERS

1 – From J. Taillet

Q – What data do you have for frequencies above 10 MHz ? – Particularly as function of time.

A – This topic is covered in "The Radio Emissions from close Lightning" by G.N. Oetzel and E.T. Pierce of the Fourth International Conference on the Universal aspects of Atmospheric Electricity, sponsored by the Joint Committee of Atmospheric Electricity of the International Association of Meteorology and Atmospheric Physics and International Association of Geomagnetism and Aeronomy by the Science Council of Japan during may 1978.

2 – From B. Burrows

Q – Why, in the NASA model, is the rise time of the first restrike so fast. Do statistics show this.

A – Recent date indicata that there is little difference in rise times for first and subsequent strokes – see Cianos & Pierce. "A Ground Lightning Environment for Engineering Usage" Technical Report SRI Project 1834 Contract L.S.-28/7-A3. Stanford Research Institute, Menlo Park, Ca. (August 1972). This information (see p. 33 of referenced report) was used in devising the NASA model.

3 – From Don Clifford

Q – How much significance should be attributed to the measurements of induced voltages on interior wires produced by nearby activity ? Can we assume that such activity would adversely affect flight computers ?

A – They are significant in that the measured current levels are comparable to those produced by the direct strike to the Learjet. Thus if a digital system is adversely affected by a direct strike, it is likely to be similarly affected by the much more numerous nearby strokes.

4 – From P.F. Little

Q – Does the continuing current in the NASA model fall to zero before the restrike ? (in real lightning there is a current zero before a restrike)

A – The NASA model does not fall to zero before the restrike, see Appendix A "Space Shuttle Lightning Protection Criteria Document" JSC 074 36 NASA L.B. Johnson Space Center, Houston (Texas) (September 1973). This model is consistent with Fig. 35 of Cianos & Pierce "A Ground-Lightning Environment for Engineering Usage".

Q – Does the Cianos & Pierce summary of Lightning characteristics present the actual time for which continuing current flows, or the interval between restrikes in which current flows ? (If the latter, the time for which current flows is very significantly less than the figures quoted by Cianos and Pierce).

A – Fig. 13 of Cianos & Pierce indicates that continuing currents flow for 200 ms in roughly 50% of all flashes, and that they flow for 300 ms in 10% of all flashes. Since the NASA model was intended to be representative of a severe flash, the 300 ms duration was chosen.

BIOGRAPHICAL NOTE

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Program Manager
Electromagnetic Sciences Laboratory
Electronics and Radio Sciences Division

SPECIALIZED PROFESSIONAL COMPETENCE

Theoretical and experimental studies of flight vehicle electrification development and test of devices for precipitation-static elimination.

REPRESENTATIVE RESEARCH ASSIGNMENTS AT STRI (since 1954)

Design of experiments and instrumentation for studies of electrostatic characteristics of large rocket vehicles and satellites.

Investigation of multipactor discharge and other breakdown mechanisms on spacecraft antennas and power systems.

Investigation of problems associated with the radiation and reception of electromagnetic signals from aircraft and guided missiles.

Studies of some effects of the nuclear electromagnetic pulse including coupling of the pulse to cable systems.

Studies of inadvertent initiation of electroexplosive devices by radio frequency and static electricity.

Studies of helicopter discharging techniques.

Studies of the role of electrostatics in contributing to spacecraft contamination, and rf interference.

Studies of characteristics of nearby lightning and its effects on aircraft electronic systems.

ACADEMIC BACKGROUND

B.S.E.E. (magna cum laude, 1951), and M.S.E.E. (1953), University of Washington ; Ph. D. in electrical engineering (1958), Stanford University.

PUBLICATIONS

Author or coauthor of several articles appearing in open literature, including "Recent Developments in Aircraft Antenna Precipitation Static", USAF Antenna Symposium Proc. (October 1968) (with E.F. Vance) ; "Internal Voltages and Currents in Solid-Shielded Cables," IEEE Symposium Proc. (July 1968) (with E.F. Vance) ; "An analysis of Corona-Generated Interference in Aircraft," and "Some Techniques for the Elimination of Corona Discharge Noise in Aircraft Antennas," Proc. IEEE (January 1964) (both with R.L. Tanner).

Author of several talks presented at professional meetings or symposia Chairman and organizer, Static Electricity section, Lightning and Static Electricity Conference, Miami Beach, Florida (3-5 December 1968), San Diego, California (9-11 December 1970) and Las Vegas, Nevada (12-15 December 1972).

PROFESSIONAL ASSOCIATIONS AND HONORS

Institute of Electrical and Electronics Engineers (senior member) –– professional group on Antennas and Propagation

Phi Beta Kappa ; Sigma Xi ; Tau Beta Pi

June 1977.

A NEW STANDARD FOR LIGHTNING QUALIFICATION TESTING OF AIRCRAFT: TECHNICAL OVERVIEW, DEFINITIONS AND BASIC WAVEFORMS

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U.S.A.

SUMMARY

The imperfect lightning-safety record of present-day aircraft, plus the appearance in new aircraft of advanced composite materials and solid state electronic systems that may be even more vulnerable to lightning strikes has led to a demand, from all quarters, for incorporation of improved lightning protection in new aircraft. This may be accomplished, for a new aircraft, in six steps which include: determination of strike zones, establishment of lightning environment, identification of vulnerable components and protection criteria, design of protective measures, and verification of their adequacy by test. The degree of protection achieved is dependent, in part, upon the lightning environment and qualification test criteria employed. In an effort to provide widely accepted criteria representative of a severe environment, Society of Automotive Engineers (SAE) Committee AE4-L (on lightning) and a similar group in UK have prepared documents defining lightning waveforms and testing techniques for aerospace vehicles and hardware. Agreement exists on the criteria to be employed for qualification testing, and the U.S. Committee has drafted a U.S. military standard for lightning qualification testing based on this criteria. This paper summarizes the scope, lightning strike zone definitions and test waveforms presented in the draft standard. Succeeding papers explain each of the qualification tests.

Background

The imperfect lightning-safety record of present-day aircraft, plus the appearance of advanced composite materials and solid state electronic systems that may be even more vulnerable to lightning strikes in the future has led to a demand for incorporation of improved lightning protection in new aircraft. This may be achieved by the following steps:

1. Determine the Lightning Strike Zones

   Determine the aircraft surfaces, or zones, where lightning strike attachment to the aircraft is probable, and the portions of the airframe through which lightning currents must flow between these attachment points.

2. Establish the Lightning Environment

   Establish the component(s) of the total lightning flash current to be expected in each lightning strike zone. These are the currents that must be protected against.

3. Identify Vulnerable Systems or Components

   Identify systems and components that might be vulnerable to interference or damage from either the direct effects (physical damage) or indirect effects (electromagnetic coupling) produced by lightning.

4. Establish Protection Criteria

   Determine the systems and/or components that need to be protected, and those that need not be protected, based upon importance to safety-of-flight, mission reliability or maintenance factors. Establish lightning protection pass-fail criteria for those items to be protected.

5. Design Lightning Protection

   Design lightning protection measures for each of the systems and/or components in need of protection.

6. Verify Protection Adequacy by Test

   Verify the adequacy of the protection designs by laboratory qualification tests simulating the lightning environments established in step 2 using the pass-fail criteria of step 4.

The lightning strike zones (step 1) are usually determined by comparison of the new aircraft with lightning strike experience of earlier aircraft of the same general shape; or by performance of simulated lightning tests on model aircraft as discussed in papers no. 8 and no. 9. The lightning strike zones of one aircraft determined from a model test (Reference 1) are shown on Figure 1.

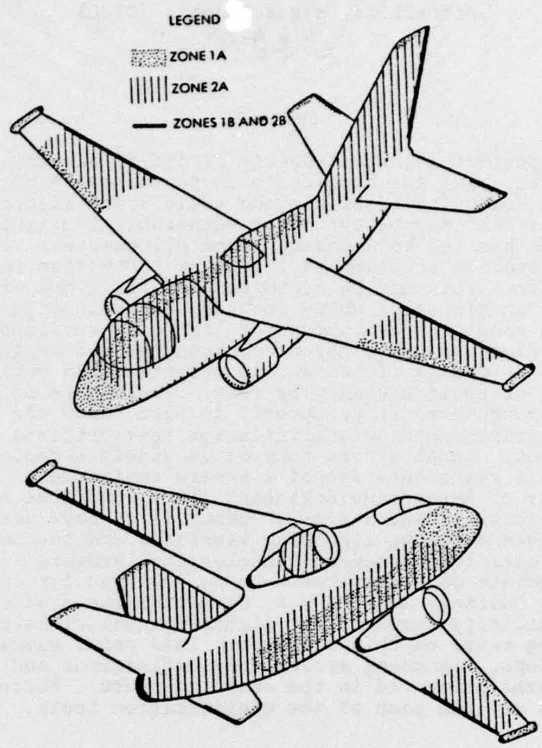

Figure 1 - Typical Lightning Strike Zones.

Until recently, in the absence of a widely accepted standard, the lightning environment expected within each zone (step 2) was designated by individual procuring organizations, or by the lightning laboratory performing the tests, with the result that the degree of protection obtained varied widely from place to place. This was particularly true in the United States, where a number of agencies and laboratories were at work designing lightning protection and performing verification tests.

Identification of potentially vulnerable systems or components (step 3) and protection criteria (step 4) has always been the responsibility of the aircraft designer, although often in consultation with regulatory authorities or lightning laboratory personnel. Design of lightning protective measures (step 5) has also been accomplished by the aircraft designers, with frequent help from specialists in lightning laboratories where potential solutions can be evaluated.

The number and severity of verification tests (step 6) were, like the environment (of step 2) also left for the procuring organization, regulatory authority or lightning laboratory to designate, with the result that laboratory capabilities (and not the natural environment) determined the severity of the tests that would be performed. Inevitably, some disagreement existed among laboratories as to what constituted an appropriate test.

The appearance in the late '60's of advanced composites and microelectronics in aircraft applications placed much greater importance on lightning protection, and emphasized the need for improved, and more widely accepted verification criteria. Prompted by this, a large amount of research was initiated into the nature of lightning and its effects upon the new composite materials and solid-state electronics. A greater understanding of lightning effects resulted, together with improved test and protective techniques. Forums such as the series of Lightning and Static Electricity Conferences beginning in 1968 (References 2, 3, 4 and 5) were created for review of results and exchange of ideas.

The need, however, for widely-accepted criteria with which to verify protection adequacy remained. Accordingly, efforts were begun in the US and the UK toward development of standardized criteria defining lightning voltage and current waveforms, severities, and test techniques for verifying the adequacy of protective measures. In the US, this effort was begun in 1972 with the formation of "Special Task F" under the auspices of the Society of Automotive Engineers (SAE) Committee AE4 on Electromagnetic Compatibility. Special Task F was comprised of 15 individuals from US lightning simulation laboratories, government agencies and aircraft manufacturers with experience in lightning simulation and testing. Beginning with a survey of the extensive literature that had become available, this committee proceeded to define the possible lightning strike zones, the lightning

environment to be experienced in each, and the techniques for simulating this environment in the laboratory. Three years later, the committee published its recommendations in a report entitled, "Lightning Test Waveforms and Techniques for Aerospace Vehicles and Hardware" dated 5 May 1976 (Reference 6). This report (sometimes called the "red book" after its cover) has been in use since 1976 by procuring organizations and lightning test laboratories in the US.

A similar effort was undertaken in the UK, culminating in publication of a "Recommended Practice for Lightning Simulation and Testing Techniques for Aircraft" by J. Phillpott of the Culham Laboratory Lightning Studies Unit in May, 1977 (Reference 7). The current waveforms described in this document have been accepted in the UK for testing aircraft systems, structures and equipment for airworthiness purposes, and are in agreement with the lightning current test criteria in the US "red book". Agreement on this important set of tests resulted from a desire to have mutually acceptable criteria and was achieved from continuing discussion among members of each group over a period of years while these documents were being formulated.

A disparity remained between the US and UK recommendations with respect to model aircraft testing, and also with respect to certain other tests utilized mainly to obtain engineering design data; areas in which some difference among US opinion remained as well. Thus, continuing discussions have been underway among both groups in an effort to resolve the remaining differences and to clarify a few ambiguities turned up during two years of use of the "red book". As a result, the US committee (now designated SAE Committee AE4-L) revised its "red book" as of June 20, 1978 and will shortly publish this document under a blue cover. This "blue book" (Reference 8) will replace the "red book" in the US.

The principal change incorporated in the US "blue book" was the establishment of two categories of tests: qualification testing and engineering testing, and inclusion of all of the protection verification (airworthiness certification) tests within the qualification test category.

The remaining tests, which included the scale model lightning attachment point test, swept stroke test and full vehicle induced voltage test, provide data useful in achieving a qualifiable design but are not considered necessary for protection verification. They are therefore placed within the engineering category. Discussion continues on the validity of several of these engineering tests, and improvements may be possible as research continues. Separation of the tests in this way eases the airworthiness authorities' task of specifying the proper qualification tests, and provides a set of tests upon which wide agreement exists.

The New Military Standard

Upon achievement of this widespread agreement on the lightning environment and test techniques for qualification purposes, the US Dept. of Defense requested SAE Committee AE4-L to draft a military standard incorporating the "blue book" qualification test criteria. This draft, entitled "Lightning Qualification Test Techniques for Aerospace Vehicles and Hardware" (Reference 9) was completed on 20 June 1978.

This new standard defines the lightning strike zones, lightning voltages and currents applicable to each zone, and the methods to be used to test components located in each zone. It is important to note that neither the standard nor the "blue book" that preceded it provide the following:

1. The location of lightning strike zones on a particular aircraft
2. The systems or components that must be tested
3. Protection techniques
4. The pass-fail criteria

These items depend upon the mission and characteristics of the particular aircraft, and should be established and agreed upon by the procuring agency, airworthiness authority and aircraft manufacturer for each new aircraft. Guidance in some of these areas will be provided in a "Users Guide" to be published as an appendix to the new standard in 1979. Further guidance is available in the literature (Reference 10).

The following are excerpts taken directly from the new standard, and discussions, where appropriate, of the definitions and lightning test waveforms incorporated in it. Discussions of each of the qualification test methods are presented in papers nos. 12 through 19.

"3.0 DEFINITIONS

3.1 Lightning Attachment Zones

Aerospace vehicle surfaces are divided into three zones, with each zone having different lightning attachment and/or transfer characteristics. These are defined as follows:

Zone 1: Surfaces of the vehicle for which there is a high probability of initial lightning flash attachment.

Zone 2: Surfaces of the vehicle across which there is a high probability of a lightning flash being swept by the airflow from a Zone 1 point of initial flash attachment."

"Zone 3: Zone 3 includes all of the vehicle areas other than those covered by Zone 1 and Zone 2 regions. In Zone 3 there is a low probability of any direct attachment of the lightning flash arc. Zone 3 areas may carry substantial amounts of electric current but only by conduction between some pair of initial or swept stroke attachment points.

Zones 1 and 2 are further divided into A and B regions depending on the probability that the flash will hang on for any protracted period of time. An A type region is one in which there is low probability that the arc will remain attached and a B type region is one in which there is a high probability that the arc will remain attached. Examples of zones are as follows:

Zone 1A: Initial attachment point with low probability of flash hang-on, such as a leading edge.

Zone 1B: Initial attachment point with high probability of flash hang-on, such as a trailing edge.

Zone 2A: A swept stroke zone with low probability of flash hang-on, such as a wing mid-span.

Zone 2B: A swept stroke zone with high probability of flash hang-on, such as a wing inboard trailing edge."

Note that the above definitions define each zone, but do not provide dimensions or other description of where the zones are located on particular aircraft. Thus the former "18 inch" rule for zone 1 (Reference 11) and similar guidelines have been omitted from the new definitions. The 18-inch dimension may be appropriate for some aircraft, but experience has shown that it is not applicable to all aircraft. Establishment of the zone locations is therefore the first step in design of protection for a new aircraft. Other definitions provided in the new standard are:

"3.2 Direct and Indirect Lightning Effects

The lightning effects to which aerospace vehicles are exposed and the effects which are reproduced through laboratory testing with simulated lightning waveforms are divided into DIRECT EFFECTS and INDIRECT EFFECTS. The DIRECT EFFECTS of lightning are the burning, eroding, blasting, and structural deformation caused by lightning arc attachment, as well as the high-pressure shock waves and magnetic forces produced by the associated high currents. The INDIRECT EFFECTS are predominantly those resulting from the interaction of the electromagnetic fields accompanying lightning, with electrical apparatus in the aircraft. Hazardous indirect effects could in principle be produced by a lightning flash that did not directly contact the aircraft and hence was not capable of producing the direct effects of burning and blasting. However, it is currently believed that most indirect effects of importance will be associated with a direct lightning flash. In some cases both direct and indirect effects may occur to the same component of the aircraft. An example would be a lightning flash to an antenna which physically damages the antenna and also sends damaging voltages into the transmitter or receiver connected to that antenna. In this document the physical damage to the antenna will be discussed as a direct effect and the voltages or currents coupled from the antenna into the communications equipment will be treated as an indirect effect."

Some of the qualification tests address direct effects, whereas others address indirect effects. These definitions clarify the intent of each.

The lightning waveforms presented in the standard are described in terms of time to crest, rate of rise, etc. Definitions of these parameters are also included, as follows:

"3.3 Waveform Parameters

Definitions of the rise time, rate of rise, decay time and other parameters utilized in the waveform definitions that follow are consistent with Paragraphs 2.4 and 2.6 of USA Standard C68.1/IEEE Standard No. 4 Standard Techniques for Dielectric Tests (1978) and High Voltage Test Techniques, IEC 60-2 (1973), Sections 4 and 6.

Average Rate of Rise Voltage

The average rate of rise, dv/dt, of a voltage waveform is defined as the slope of a straight line drawn between the points where the voltage is 30% and 90% of its peak value, as shown in Figure 3-1."

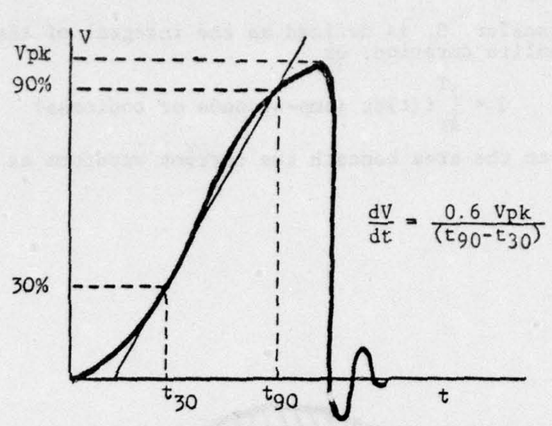

Figure 3-1 Average Rate of Rise of Voltage

Time to Crest

The time to crest, T_1, of a voltage waveform is defined as 1.67 times the time interval between the instants when the voltage is 30% and 90% of its peak value as shown in Figure 3-2.

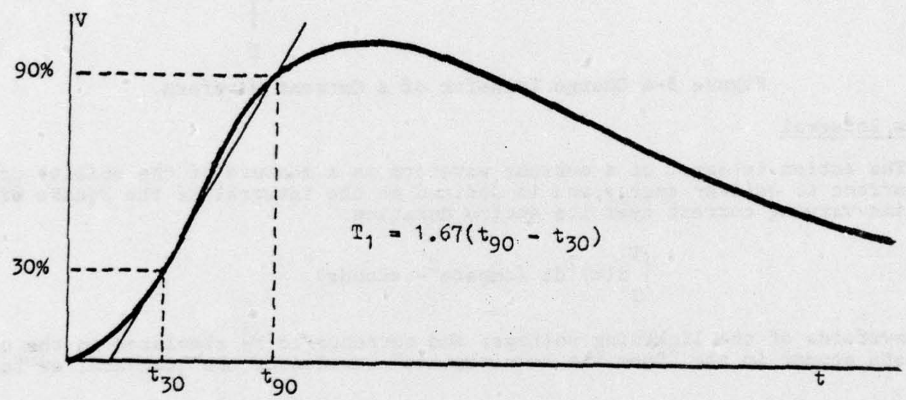

Figure 3-2 Time to Crest of a Voltage Waveform

Decay Time

The decay time, T_2, of a voltage waveform is defined as the time interval between the intersect with the abcissa of a line drawn through the points where the voltage is 30% and 90% of its peak value during its rise, and the instant when the voltage has decayed to 50% of its peak value, as shown on Figure 3-3.

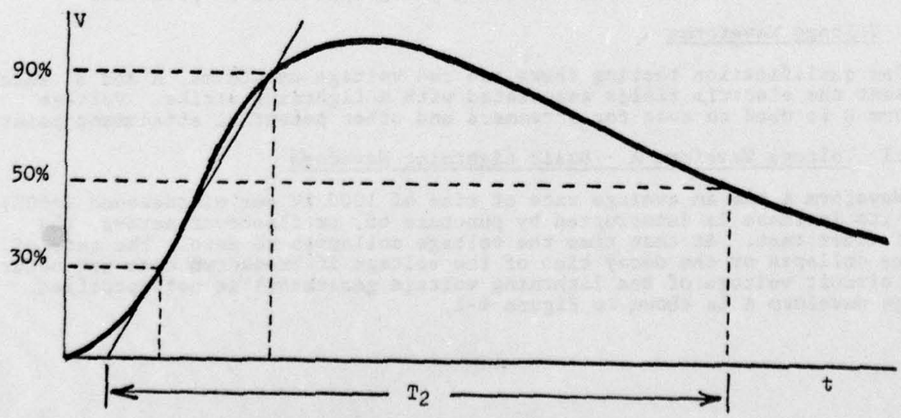

Figure 3-3 Decay Time of a Voltage Waveform

"Charge Transfer

The charge transfer, Q, is defined as the integral of the time-varying current over its entire duration, or

$$Q = \int_0^T i(t)dt \text{ (amp-seconds or coulombs)}$$

and is equivalent to the area beneath the current waveform as shown on Figure 3-4.

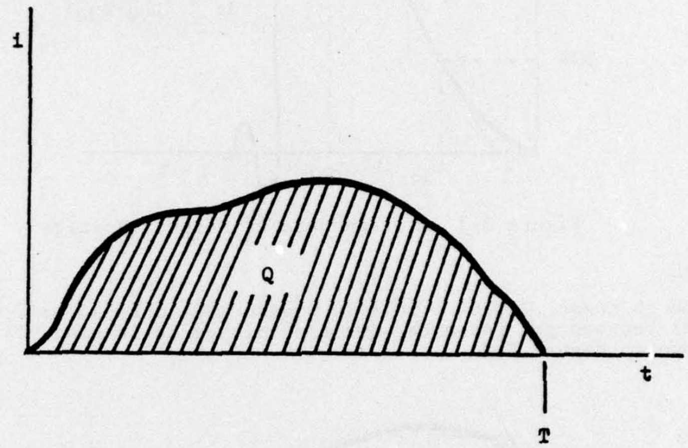

Figure 3-4 Charge Transfer of a Current Waveform.

Action Integral

The action integral of a current waveform is a measure of the ability of the current to deliver energy and is defined as the integral of the square of the time-varying current over its entire duration.

$$\int_0^T i(t)^2 dt \text{ (ampere}^2\text{-seconds)"}$$

The waveforms of the lightning voltages and currents to be simulated in the qualification tests appear in the "Specific Requirements" section of the standard, as follows:

"4.2 Specific Requirements

Waveforms of the simulated lightning currents and voltages to be used in these tests are presented in this section.

4.2.1 Test Waveforms

The waveforms and components depicted in Figures 4-1, 4-2 and 4-3 are idealized representations and need not be simulated exactly. Only the numerical parameters specified in the following paragraphs need be produced.

4.2.2 Voltage Waveforms

For qualification testing there are two voltage waveforms, A and B, which represent the electric fields associated with a lightning strike. Voltage waveform B is used to test for streamers and other potential attachment points.

4.2.2.1 Voltage Waveform A - Basic Lightning Waveform

Waveform A has an average rate of rise of 1000 kV per microsecond (+50%) until its increase is interrupted by puncture of, or flashover across, the object under test. At that time the voltage collapses to zero. The rate of voltage collapse or the decay time of the voltage if breakdown does not occur (open circuit voltage of the lightning voltage generator) is not specified. Voltage waveform A is shown in Figure 4-1."

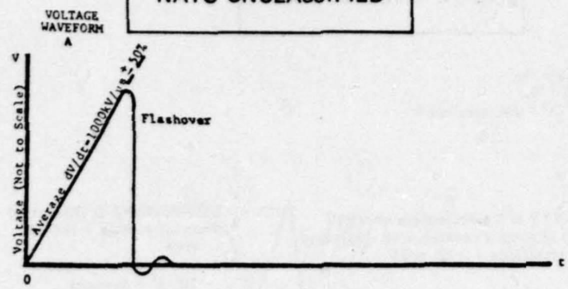

Figure 4-1 Voltage Waveform A

4.2.2.2 Voltage Waveform B - Full Wave

Waveform B rises to crest in 1.2 (±20%) microseconds. Time to crest and decay time refer to the open circuit voltage of the lightning voltage generator, and assume that the waveform is not limited by puncture or flashover of the object under tests. This waveform is shown in Figure 4-2.

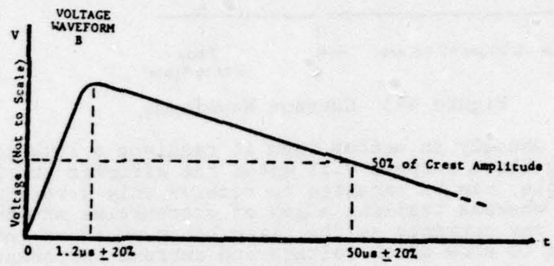

Figure 4-2 Voltage Waveform B

4.2.3 Current Waveforms and Components

For qualification testing, there are four components, A, B, C and D, used for determination of direct effects, and test waveform E used for determination of indirect effects. Components A, B, C and D each simulate a different characteristic of the current in a natural lightning flash and are shown in Figure 4-3. They are applied individually or as a composite of two or more components together in one test. Current waveform E, also shown on Figure 4-3, is intended to determine indirect effects.

4.2.3.1 Component A - Initial High Peak Current

Component A has a peak amplitude of 200 kA (±10%) and an action integral ($\int i^2 dt$) of 2×10^6 amp^2-second (±20%) with a total time duration not exceeding 500 microseconds. This component may be unidirectional or oscillatory.

4.2.3.2 Component B - Intermediate Current

Component B has an average amplitude of 2 kA (±10%) flowing for a maximum duration of 5 milliseconds and a maximum charge transfer of 10 coulombs. The waveform shall be unidirectional, e.g. rectangular, exponential or linearly decaying.

4.2.3.3 Component C - Continuing Current

Component C transfers a charge of 200 coulombs (±20%) in a time of between 0.25 and 1 second. The waveform shall be unidirectional, e.g. rectangular, exponential or linearly decaying.

4.2.3.4 Component D - Restrike Current

Component D has a peak amplitude of 100 kA (±10%) and an action integral of 0.25×10^6 amp^2-second (±20%). This component may be either unidirectional or oscillatory with a total time not exceeding 500 microseconds.

4.2.3.5 Current Waveform E - Fast Rate of Rise Stroke Test for Full-Size Hardware

Current waveform E has an instantaneous rate of rise of at least 25 kA per microsecond for at least 0.5 microsecond, as shown in Figure 4-3. Current waveform E has a minimum amplitude of 50 kA. Alternatively, components A or D may be applied with a 25 kA per microsecond rate of rise for at least 0.5 microsecond and the direct and indirect effects evaluation conducted simultaneously."

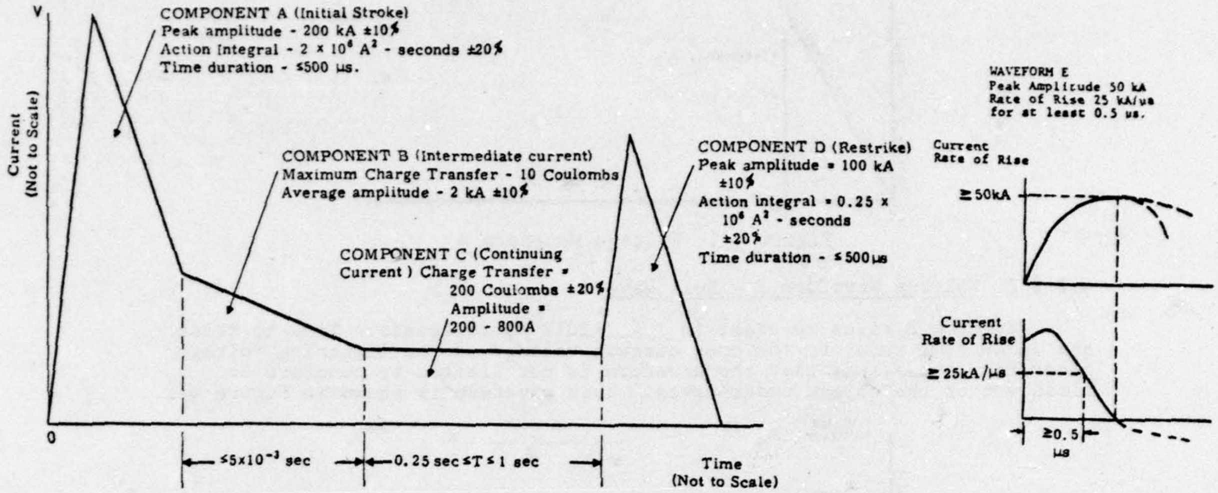

Figure 4-3 Current Waveforms

Since the aircraft is usually in motion when it receives a lightning strike, not all components of the lightning flash current will enter the aircraft at the same point. Surfaces in zone 2A, for example, can be expected to receive only a re-strike and a portion of the continuing current, whereas trailing edges of extremities may be struck initially and have to receive all of the currents as the flash hangs on there until it dies. Accordingly, a table is presented to show which voltage and current components are required in each zone:

Table I - Application of Waveforms for Qualification Tests

Test	Zone	Voltage Waveforms A	Voltage Waveforms B	Current Waveforms/Components A	B	C	D	E	Test Method
Full size hardware attachment point	1A,B	X							102
Direct effects-structural	1A			X	X				301
"	1B			X	X	X	X		301
"	2A				X[1]	X[1]	X		301
"	2B				X	X	X		301
"	3			X		X			301
Direct effects-combustible vapor ignition		Same current components as for structural tests							302
Direct effects-streamers			X						303
Indirect effects-external electrical hardware								X[2]	401

Note 1: Use an average current of 2 kA $\pm$ 10% for the actual dwell time up to 5 msec, and an average current of 400 amps for the remaining dwell time.

Note 2: Indirect effects should also be measured with current components A, B, C or D as appropriate to the test zone."

The test methods (identified by number 102, 301, etc.) in which these waveforms are applied are also listed in Table I. The objectives of each test, along with setup, measurement, and data requirements are described in succeeding parts of the standard and in papers nos. 12 through 19.

Severity

Current Components A, B, C, D and E, taken together represent a severe lightning flash, one that is exceeded in action integral only about 1% of the time (Reference 12). It is akin to the "Severe (applied) model" of a lightning flash proposed by Cianos and Pierce (Reference 13). The statistical data against which this comparison is made, of course, was derived from measurements of lightning currents entering grounded objects. There is some reason to believe that, statistically, current amplitudes and action integrals would be somewhat lower at flight altitudes due to branching and higher source impedances. If this is so, the probability of encountering a flash of greater severity than the new-standard criteria would be even less than 1%.

Superseded Documents

If adopted by the US Dept. of Defense, the lightning test criteria in the standard will supersede those in present US military procurement specifications and test standards. These include:

MIL-B-5087B (ASG) - Bonding, Electrical, and Lightning Protection for Aerospace Systems

MIL-A-9094D (ASG) - Arrester, Lightning General Specification For

MIL-C-38373A (ASG) - Cap, Fluid Filler

Future editions of these and specifications for other hardware in need of lightning tests will make reference to the new standard for lightning tests. It is expected that a similar situation will apply to US Civil (FAA) airworthiness certifiction requirements.

Conclusion

Preparation of the "blue book" and "green book" criteria for lightning testing of aircraft, achievement of widespread US and UK agreement on its validity, and preparation of the new standard that resulted from it represent a significant advancement in the resources available to provide reliable lightning protection for aircraft. This advancement did not come about quickly nor was it accomplished by only a few individuals. It was instead the product of a cooperative effort of many individuals on both sides of the Atlantic. This author wishes to thank all who participated in this effort, and looks forward to continued cooperation in the future.

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9. "Lightning Qualification Test Techniques for Aerospace Vehicles and Hardware", proposed military standard, SAE Committee AE4L, 20 June 1978.

10. F.A. Fisher and J.A. Plumer, "Lightning Protection of Aircraft", National Aeronautics and Space Administration Reference Publication 1008, October 1977.

11. "Protection of Aircraft Fuel Systems Against Lightning", Federal Aviation Advisory Circular No. AC 20-53.

12. N. Cianos and E.T. Pierce, "A Ground-Lightning Environment for Engineering Use", Technical Report 1, prepared by the Stanford Research Institute for McDonnell Douglas Company, August 1972, p. 91.

13. Cianos and Pierce, pp. 82-93.

QUESTIONS and ANSWERS

1 – From B. Burrows

Q – Are test levels based on 2% or 1/2% severity levels ?

A – The 200 kA amplitude of test current component A was selected by the US SAE committee to represent a 2% probability stroke, based on statistics of peak currents compiled by Cianos and Pierce in Reference 12 of the paper. This means that 98% of all strokes are expected to be less severe than this and 2% may be more severe, in terms of amplitude. The action integral of this component (2×10^6 A^2s) was set higher than that of a 2% severity return stroke (1×10^6 A^2s) to account for higher action integrals that sometimes occur with positive polarity strokes.

The intermediate current (component B – 2 kA), continuing current (component C – 200 coulombs), and re-strike current (component D – 100 kA at 0.25×10^6 A^2s), and rate-of-rise test (component E – Scaled to 100 kA/µs), also represent approximately 2% severities.

2 – From B.L. Perry

Q – The zones shown in Fig. 1 as "Typical Lightning Strike Zones" are not, in our experience, at all typical. Why therefore use such a figure as an illustration - it could by very deceiving ?

A – The lightning strike zones of Figure 1 resulted from the model attachment test reported in Reference 1 of this paper. The zones shown aft of leading edges were included to represent swept strokes. With the exception of the zones pictured in the middle of the wing, all of the other zones shown in the figure are indeed typical of in-flight lightning attachments on a great many aircraft.

The mid-wing zones resulted from a small number of strikes to the leading edges of the model wings ; the result, perhaps, of the unusual length of this wing having reduced the shielding afforded by the fuselage. The test was made with a high voltage rising to its crest in 50 microseconds, illustrating the tendency of slow-rising waveforms to bring out less probable (but in this case quite important) attachment zones.

3 – From G. Odam

Q – Bearing in mind that effective forces due to magnetic effects depend on the relationship of the mechanical resonant frequency of the system to the waveform rise time and decay time constant and particularly the decay time constant, as well as peak current, which waveform should be used to simulate magnetic force effects without risk of over design ?

A – Current component A (200 kA, 2×10^6 A^2s) should be used to simulate magnetic force effects because its amplitude, action integral and time duration are each important in determining the degree of mechanical force damage that occurs.

4 – From S.D. Schneider

Q – (Comment) : 1. Your slide on steps in lightning protection development shows "Step 6, verify adequacy by test", of course, analysis may be conducted instead in some cases to show adequacy. Testing may NOT be required.

A – Concur, where sufficient confidence exists in the analytical techniques used.

Q – Was the leading edge of the model (in Figure 1) smooth (i.e. no leading edge devices extended) ?

A – Yes. The leading edges were smoothly rounded.

BIOGRAPHICAL NOTE

J. Anderson PLUMER

President – Lightning Technologies, Inc.

EDUCATION

BS in Electrical Engineering, Lehigh University, 1961.

EXPERIENCE

Mr. Plumer has spent most of his engineering career in the study of lightning hazards and development of protection methods for aircraft and ground installations. He began this work in 1965 after joining the General Electric High Voltage Laboratory. At the High Voltage Laboratory he conducted research into lightning effects on aircraft fuel systems, advanced composites, electrical systems and flight controls under projects supported by the Dept. of Defense, NASA, FAA and industry. He was instrumental in the development of several new test techniques for evaluation of lightning effects on aircraft, including the Lightning Transient Analysis (LTA) technique for measurement of lightning-induced voltages in aircraft electrical circuits, and was inventor of the GE Aircraft Lightning Suppressor which won an IR-100 award in 1976. In 1977 Mr. Plumer founded Lightning Technologies, Inc. to apply the results of this research more directly to the lightning protection design, consulting and certification testing needs of industry and government organizations.

Mr. Plumer is co-author of the book, "Lightning Protection of Aircraft", published in 1977 by the National Aeronautics and Space Administration ; and the author of numerous technical reports, papers and articles on subjects related to lightning. He has established pilot reporting projects to identify potential lightning hazards to transport and general aviation aircraft, and contributed to industry groups working toward improved standardization of lightning qualification testing.

Mr. Plumer is co-chairman of the Society of Automotive Engineers Committee AE4L on Lightning Testing of Aircraft ; a Senior Member of the Institute of Electrical and Electronics Engineers, and past chairman of its Berkshire Section.

LABORATORY TESTS TO DETERMINE LIGHTNING ATTACHMENT POINTS WITH SMALL AIRCRAFT MODELS (ENGINEERING TEST)

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SUMMARY

The simulation of lightning strikes to model aircraft involves an extrapolation from the behaviour of laboratory sparks to fullscale lightning. This is not possible in detail, particularly since an isolated model is interposed into the two-electrode gap normally studied in the laboratory. This paper compares the attachment process to an aircraft with laboratory experiments, and discusses the influence of gap size, geometric scaling and voltage waveshape on the spread of attachment points. Particular emphasis is laid on the identification of low-probability attachment areas. A comparison with flight experience is the best test of the validity of any test technique, but no clear and complete agreement exists as to the best approach. Parameters and procedures that are generally acceptable are noted.

1. INTRODUCTION

High-voltage testing for ground installations was an established technique before the first attempts to simulate lightning strikes to aircraft were made. The electrical power industry was forced to develop high-voltage testing technology for distribution networks and also to simulate the effects of lightning strikes to these distribution systems. The means of producing several megavolts in the laboratory was at hand, making possible the use of sparks several metres long. Thus aircraft models about 1 m long could be placed in the path of a long spark and the attachment points noted.

If valid test techniques can be established the relative probability of first strikes to various parts of any aerospace vehicle may be estimated from model tests. The change of scale dimensionally is about 50:1, but the pressure is unchanged (if low altitude flight is considered) or at most increased by 2:1. There are additional differences in that the aircraft velocity is not represented, and the pressure differences around it are not reproduced. This velocity is not likely to be important, since the leader stroke advances at speeds of 10^5 m/s to 10^6 m/s compared to the aircraft speed of less than 1000 m/s even for supersonic flight. Pressure differences affect the ease with which corona discharges can appear, and this could be important (see next section).

Laboratory studies of arcs between 1 m and 25 m in length show many general features independent of length (Allibone 1977). The processes of ionisation and breakdown cannot be scaled down merely by scaling dimensions, but some comparability exists. The validity of model tests, however, can only be demonstrated satisfactorily by comparing test results with flight experience of the actual aircraft. Retrospective testing of models of aircraft with long service records offers one way of evaluating test techniques. For commercial aircraft that have been in operation for many years, considerable information on the position of strike points exists (Perry 1968, 1970; Plumer 1972; Plumer and Perry (1975).

For an aircraft of the same general shape as those which have yielded this data, the probable strike points can be predicted with fair certainty. The initial strikes occur to nose, tail and wing tips, with swept strokes usually along the fuselage. Model tests are not necessary for such aircraft. Any unusual shape of aircraft introduced would benefit from model tests, if a satisfactory technique exists.

It is not always essential to derive the relative probabilities of initial attachment to all parts of the aircraft from work on models. The interest often lies in identifying areas of very low probability of attachment. Structures or components vulnerable to lightning damage are located in Zone 3 areas on the aircraft, and these zones must be known. Identification of Zone 3 areas and an assessment of the probability of a strike to the general region of each area is the most usually demanded from model tests. Obviously techniques which are satisfactory in meeting these limited objectives may not be successful in predicting relative probabilities elsewhere on the aircraft surface.

If it is required to determine the probability of attachment to a small structure, it is wise to supplement model tests (which provide an indication of probability for the region around the structure) with tests on full-scale items. The surrounding area should be modelled in appropriate detail.

In order to make model testing as realistic as possible the behaviour of natural lightning must be considered. This paper discusses the initial attachment process, and then considers the bahaviour of long laboratory arcs in order to determine the geometry preferred for the test configuration. The voltage waveform is then discussed on the basis of

laboratory experience, including studies aimed at obtaining a wide spread of attachment points on a simplified model. Test procedures and parameters that are generally accepted are described, and remaining problems indicated.

2. LEADER ATTACHMENT TO AIRCRAFT

In principle an aerospace vehicle could be struck:

a. by flying into an existing lightning channel

b. by initiating a flash

c. by becoming part of a developing leader channel.

The probability of (a) is extremely small, because the duration of a flash is less than one second and the aircraft travels only a short distance in this time. The size of the aircraft is small compared to the length of lightning channel to ground, so that the probability of (b) is also small except for cloud flashes (Schaeffer 1972) which are of lower current. Experiments on triggered lightning (Gary and Fieux 1975) using rockets trailing wire have shown that even in favourable circumstances distances of about a hundred meters must be bridged to initiate a discharge. Very large rockets produce exhausts of sufficient length and conductivity to make it likely that they might trigger lightning, as in the Apollo 12 incident. (Fisher and Plumer 1977, Chapter 2: Pierce 1972) Jet trails from aircraft are not sufficiently conducting to act in this way (Schaeffer 1972). Thus only (c) remains to be considered.

Figure 1 shows the process by which a leader approaching an aircraft from a cloud at

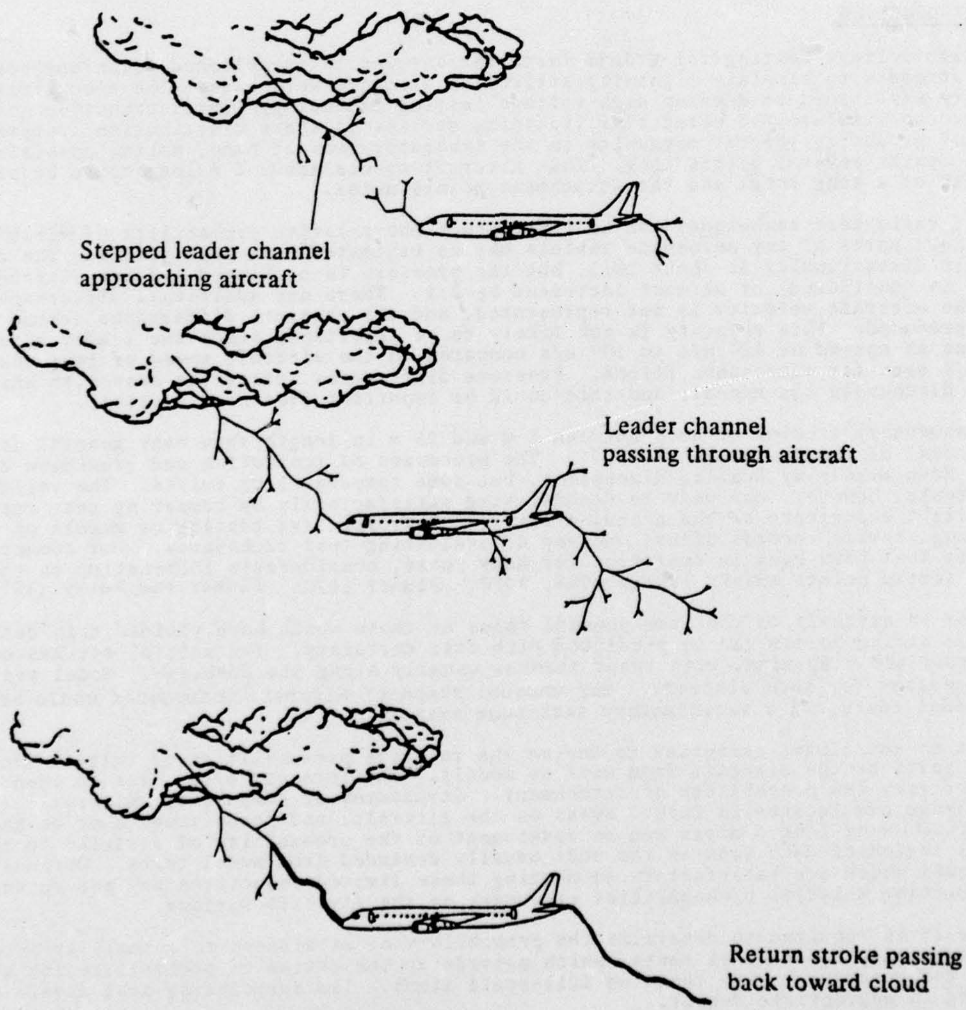


Figure 1 - Lightning flash striking an aircraft

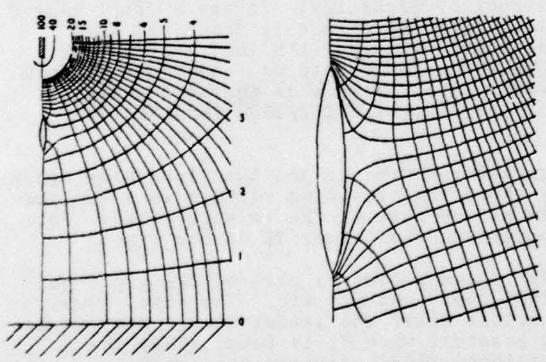
high potential induces high electric fields at extremities of the aircraft. This electric stress becomes high enough for corona discharges to appear, and eventually from these

regions leaders emerge from the aircraft. Pressure changes in flight may affect the corona on-set, and so alter the probability of attachment somewhat. Leaders from cloud and aircraft show branching, and when the branching channels are close enough they merge. The aircraft is now part of the leader channel, which continues to grow. In the illustration the most dangerous type of flash is shown, a cloud-to-ground discharge where a return stroke follows the leader. Here we are concerned only with the initial attachment process, which is similar for cloud or ground flashes.

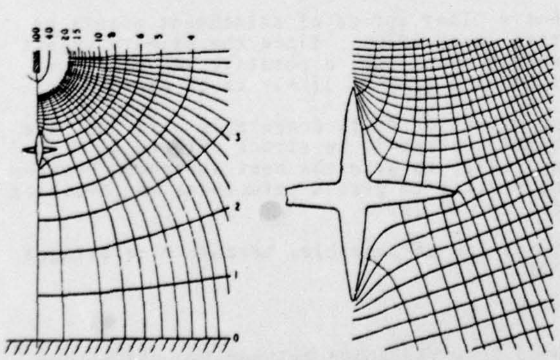
The average velocity of the approaching leader as measured photographically is between 1 and 8×10^5 m/s. The leader appears to move in steps of average length 25 m but with a wide range from 10 to 200 m, with pauses between steps of about 50 μs (range 10 to 100 μs). These figures are well established (Berger 1977). The luminous front advances at a speed of 5×10^7 m/s during one step, and it is believed (Uman 1969, Chapter 7) that this rapid advance is possible only because some form of preionisation has taken place ahead of the luminous front. This process is generally assumed to be continuous, because the electrostatic field changes associated with the developing stepped leader are relatively smooth (Schonland 1953). The implication is that charge moves continuously, even though the luminous channel shows intermittent brightening, with a very bright new step added at the tip every time the channel is luminous.

The aircraft then is approached by a leader of radius 1 to 10 m (Uman 1969) whose tip is moving at 10^5 to 10^6 m/s. This is equivalent to an extending conductor of high potential with respect to ground, typically about 50 MV.

It is of great interest to determine the time scale on which the electric field at the aircraft surface rises. The three-dimensional problem is soluble in principle, but results so far reported have been only two-dimensional treatments (James and Philpott 1971). Figure 2 shows two approximations, a cigar-shaped 'fuselage' without wings, and a cigar-plus-disc model to obtain an upper limit for the effect of wings in reducing the field at the tip of the fuselage. On the assumption that the charge on the leader is 5×10^{-4} C/m (Berger 1977) and that the radius of the leader is 5 m the potential at 500 m distance is 33 MV. With decreasing distance thus potential changes only slightly at first, assuming that the aircraft remains uncharged. This is a good approximation, because the aircraft has a large number of sharp-pointed conductors at various points over its surface which will discharge easily. Figure 3 shows the variation of voltage and stress

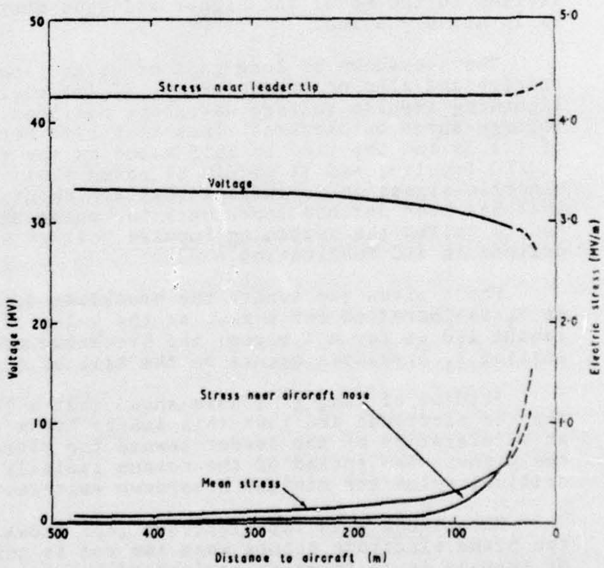


Electric field distribution for a leader approaching an aircraft neglecting the wings.

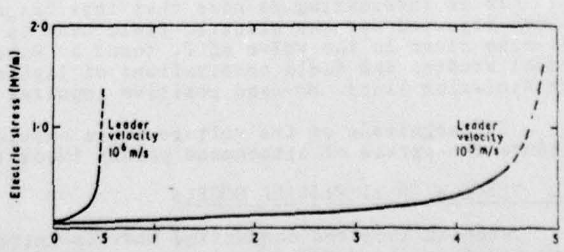


Electric field distribution for a leader approaching an aircraft assuming axial symmetry.

Figure 2



Electric stress and voltage between leader and aircraft against distance



Mean electric stress against time for various leader velocities.

Figure 3

with distance and time, assuming extreme values for the leader speeds. Significant charges in the mean stress in the gap and in the stress at the nose appear when the separation is about 100 m. The mean stress doubles in a distance of about 30 m, or in a time between 30 and 300 μs for the fastest and slowest leaders.

The absolute value of the stress will depend on the size of the aircraft and on the potential of the streamer. An electric field E of about 2.5 MV/m at a conductor surface is necessary before breakdown can occur; above this value ionisation overcomes attachment and electron multiplication is possible. A minimum volume of air in which $E > 2.5$ MV/m is also required to sustain corona, because a succession of electron avalanches must be possible. From the figure the critical value of E will occur at the nose when the leader is 20 m distant. If the range of leader charge values is 2 to 20×10^{-4} C/m (Fisher & Plumer 1977) the range of values for the gap when corona can start at the nose is 10 to 50 m approximately.

An alternative way of assessing the striking distance is used in considering the protection of buildings from lightning. Golde (1977) uses the concept of a critical field across the final air gap. If the mean stress is greater than 500 kV/m for a negative leader or 300 kV/m for a positive leader then attachment will occur. From Figure 3 this criterion gives a striking distance of about 50 m for negative leaders and 70 m for positive instead of 20 m. The range of striking distances for leaders of different charge density is then 25 to 120 m approximately for negative strokes and even greater for the rarer positive strokes.

It is clear from Figure 3 that the field structure close to an aircraft will be dominated by the aircraft's shape. The approaching leader is large, 5 m in radius at a distance of at least 10 m when corona starts. The equipotentials around the leader, therefore, have large radii of curvature at all positions except close to surface of the aircraft. Thus to obtain an appropriate corona distribution around a model the equipotentials in the test must also be of large radius on the appropriate scale.

3. DEVELOPMENT OF THE BASIC TEST CONFIGURATION

Since it is desirable to encourage corona discharges at the model surface, the test electrodes themselves should ideally be of large radius compared to the radii of curvature in the model. One indeed can conveniently be flat, the earthed floor of the laboratory. If the other is also made flat, it becomes difficult to obtain a reasonable fraction of strikes to the model and higher voltages must be applied. Thus the high-voltage electrode is invariably a rod.

The breakdown of long gaps of given geometry depends on the waveform of the applied voltage and also on the polarity, if the electrodes are of different shape. A standard lightning impulse voltage waveshape has been agreed internationally to represent the voltage surge on overhead lines that have been struck by lightning. It has a front time T of 1.2 μs and the time to half value on the tail T_2 is 50 μs. This is described as a 1.2/50 impulse, and it should be noted that is has nothing to do with the rise-time of electric stress on conductors that are about to be struck by lightning. Another waveform that has been defined represents the surge on transmission lines when they are switched and is called the switching impulse voltage waveshape. It is a 250/2500 impulse (as defined in IEC Publication 60-2).

For a given gap length the breakdown voltage first decreases and then increases again as T_1 is increased for 1 μs. At the value of T_1 where the breakdown voltage is a minimum (about 150 μs for a 3 m gap) the breakdown occurs at the peak of the voltage pulse. For smaller T_1 breakdown occurs on the tail of the pulse and for longer T_1 on the rise.

Studies of long gaps have shown that a leader channel crosses part of the gap from the rod electrode and that this leader has a corona region at its tip. The final jump, an acceleration of the leader toward the plane, occurs after the leader corona has reached the plane. The spread of the corona radially is greatest when T_1 is long, above the critical value for minimum breakdown voltage. Allibone (1977) reviews this work.

Experience with two-electrode gaps shows that a wider spread of attachment points on the plane electrode occurs when the rod is positive (Meek 1970). Since the primary object of testing is to identify regions of low attachment probability, a positive rod-plane gap with a slow-rising pulse (T_1 greater than 150 μs for a 3 gap) is likely to be appropriate.

It is interesting to note that this range of values of T_1 is comparable with the rise times expected for the electric field near an aircraft about to be struck (Figure 3). It is also close to the value of T_1 found by Rühling (1972) to give the best agreement between model studies and field observations of lightning strikes to ground structures and overhead transmission lines. He used positive impulses of 125/1000 μs.

The magnitude of the voltage pulse should be as low as possible, because overvoltages reduce the spread of attachment points (Meek 1970).

4. TESTS WITH SIMPLIFIED MODELS

When an isolated conducting body is introduced into the space between two electrodes, the electric field distribution is modified and the breakdown process may be affected. Schaeffer (1972) investigated the effect of a small metal sphere placed between large, near-plane electrodes. Computations showed that the surface E-field at the sphere was always very large. However, only when the gap between the sphere and an electrode was less

than the sphere radius did the d.c. breakdown voltage fall. He concluded that to increase the electric field at an isolated conductor did not allow easier breakdown. It is necessary that some source should supply charge freely into the region of increased field.

The spread of attachment points over the sphere was not studied. It would be interesting to know how the spread changed with the position of the sphere; Phillpott et al (1975) examined this with a more complex model which still possessed cylindrical symmetry. E-fields were computed for the initial conditions (without space charge) when a disc with a rounded rim carrying a short cylindrical post was placed between the rod and plane electrodes of a 2 m and a 4 m gap. Two different pulsed voltages were applied: their waveshapes were 1.2/40 μs and 150/1600 μs.

The conclusions were:

(i) The position of the model in the gap is critical. To obtain a wide spread of attachment points the model should be isolated, not grounded.

(ii) Discharges across the lower gap restrict the spread of attachment. This gap should be greater than or equal to 2.5 d, where d is the largest model dimension.

(iii) The leader from the electrode must not dominate in the upper gap. The upper gap should be greater than or equal to 0.4 L, where L is the gap separation.

(iv) A charge in scale of 2:1 for gap and model together produced large differences in the spread of attachments, except for negative long fronted waves (150/1600 μs).

(v) Long-fronted waves give a wider spread of attachment points.

(vi) The voltage applied should be such that there is a 90% probability of flashover (V_{90}) in order to obtain data reasonably quickly.

The recommendation (iii) implies that the upper gap should be at least 1.9 d, if the model depth is about 0.3 d. The total gap is then 4.7 d, or 5.5 d if the model is rotated (as in model aircraft attachment tests) to include its greatest dimension within the gap. In view of the lack of scaling of corona with distance, it seems best to use as large a scale model as feasible.

Unwanted discharges in the lower gap must be guarded against. It is well to support the model by insulating tapes which lie approximately along equipotential surfaces close to the model. Insulating posts beneath may lead to surface tracking.

It should be noted that with a long lower gap the attachments beneath the disc are likely to be representative of attachments from long channels. With short lower gaps the attachments beneath are dominated by the presence of the lower electrode, and Schaeffer's results with the mid-gap sphere (1972) suggest that with a lower gap of 0.5 d the whole breakdown process is affected. Attachments below are probably not valid unless the lower gap is much greater than this.

The computations emphasised the importance of small charges on the model, which cause large changes in its potential because it has very small capacity. The change in the E-field distribution over the surface is very great, and the E-field near the rim can be reversed due to charging. The theory assumes cylindrical symmetry, and ignores the presence of corona or leaders from the rod. In practice asymmetrical fields will occur.

5. TESTS WITH MODEL AIRCRAFT

5.1 Test Geometry

The normal arrangement is a rod-plane gap with a model suspended near the centre of the gap. By rotating the model about the three cartesian axes the relative position of the rod with respect to the model takes the positions shown in Figure 4 if rotation steps of 15° are used. An additional plane at 45° to the yaw and pitch planes is sometimes added, giving a total of 66 positions (Stahmann 1970). Sometimes steps of 30° separation are used to reduce the cost of testing (Knoller and Plumer 1975, Phillpott et al 1975).

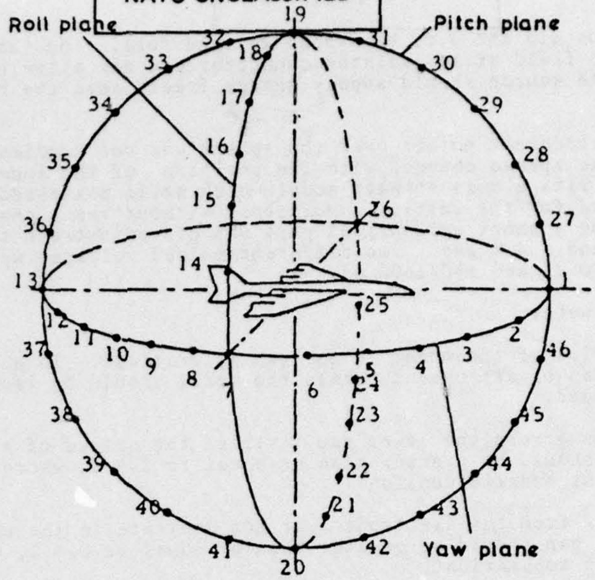


Figure 4 - Rod positions relative to model

The number of discharges from each position is usually about 10.

The scaling and upper gap spacing is largely beyond dispute, though there is some divergence about the lower gap spacing. It is largely agreed that tests should be carried out with the model isolated from ground, but some authors accept a small lower gap as giving good isolation while others use a large gap. The table shows typical choices and reasons: d is the largest model dimension, L is the gap length.

Table 1 - Geometry of Model Tests

Author	Upper Gap	Reason for Choice of Gap	Lower Gap	Model Scale
Stahmann 1970	1.5d	Results same as longer gaps		1:40
James and Phillpott 1971	1.5d	Lightning leader	-	1:30
Phillpott et al 1975	0.4L (1.9d)	Laboratory leader length	2.5d	1:72
Clifford 1975	1.5d	Lightning leader striking distance	1.0d	1:20/30/72
Knoller and Plumer 1975	2.5 m (2.4d)	Lightning leader striking distance	2.5 m (2.4d)	1:20

The changes in scale from 1:20 to 1:72 gave no significant difference in the work reported by Clifford. The accuracy of the models varied, but this appears to be unimportant. An accuracy of ± 0.5 mm in a model 0.3 m long is acceptable in practice.

Generally the whole surface of the model is conducting, with no account taken of insulating sections such as canopies or radomes. Conducting paint or foil over the surface is sufficient.

5.2 Test Voltage Waveshape

The wider spread of attachment points observed with long wavefronts in two-electrode gaps was confirmed for attachment to their simple model by Phillpott et al (1975). Work with a 1:72 scale model of a BAC 1-11 in which attachment to low-probability areas was studied was reported in the same paper. It was concluded that changing the waves shape from 1/50 μs to 85/3800 μs produced no significant effect on the spread, in disagreement with the simple model results. Further, studies of the general attachment point distribution with BAC 1-11 and HS Trident models consistently underestimated the attachment probabilities to low probability areas (compared to flight experience) for both waveshapes.

These authors referred to some earlier work by Golde et al (1971) with a 1:72 model of a HS Trident. This showed that 1 to 2 μs short-fronted waves gave attachment points at the front of the tailplane fairing which were not seen in flight. 200 μs long-fronted waves did not show these attachments, that is, they showed a narrower spread of attachments. A change in waveshape seems not to produce consistent results even with the same model airfraft.

Reynolds (1975) reported work on 1:72 models of a BAC 1-11 and HS Trident with upper gaps from 0.35 to 1.4 m and with positive and negative voltages from a 2.8 MV impulse generator. He used 1/50 μs and 85/3800 μs waveshapes at 20% above the V_{50} level. Rod

positions were spaced at 30° intervals in the roll, pitch and yaw planes, with another added plane parallel to the roll plane and 30° forward of it from the model. The strikes were divided into Axial (nose, tail, fuselage) and Transverse (wingtips, leading and trailing edges, nacelles). Table 2 shows the results compared to flight experience reported by Perry (1970) and from Lockheed flight reports (1964).

Table 2 - Strikes to BAC 1-11 & H S Trident

Source	% Distribution of Strikes Axial	Transverse
Model Tests	67	33
Flight Experience		
(i) Perry	74	26
(ii) Lockheed	70	30

From the data Reynolds deduced that one strike in 770 would go to a wing leading edge on these two aircraft, or approximately 1 in 1.6×10^6 flying hours, on a pessimistic basis. The tests actually suggest that 1 in 6×10^6 hours would be a better estimate, which fits the observation of zero strikes in 4×10^6 hours.

These results are satisfactory, but the conditions under which they were obtained encompass a wide range of conditions and involved a test series of 1300 shots in all. It would be interesting to have a more selective test report.

Clifford (1975) in the course of a wide-ranging discussion of model testing quoted results obtained in tests on the F4 fighter aircraft. These tests used a 1.5/40 μs wave, polarity not given. A theoretical argument in favour of a short-fronted wave was presented on the assumption that the leader velocity increased over the last jump. The experimental results are compared with flight experience in Table 3.

Table 3 - Strikes to F4 fighter

	Nose	Wing tip	Horizontal Stabiliser	Vertical Stabiliser	Fuselage
Model Tests (geometrical %)	33.9	32.5	12.7	11.1	9.8
Flight Experience (% of Total Strikes)	33.0	32.0	14.4	11.3	9.3

The detailed agreement is extremely good. More information on the test techniques would be valuable for comparison. Clifford commented that the influence of the generator impedance is often ignored, and that the overvoltage applied is important.

The pulse voltage waveshape and overvoltage applied need a great deal of study before any firm recommendations can be made on model testing. The influence of the generator characteristics also must be considered.

5.3 Streamering Tests

Stahmann (1970) reported that by using an arrested streamer discharge the regions of low attachment probability can be identified in a few discharges only. Then the number of positions of the electrode from which attachment to these low-probability areas is possible can be reduced to about 3. This assumes that only one small region of the circumscribed sphere containing the probe positions can give strokes to each low - probability area. Reynolds (1975) found that five regions of the sphere could give strokes to the wing root area of a Concorde model.

If all the regions that can provide strokes to the area of interest on the model can be identified, the streamering test can be useful as a quick way of selecting the areas of low attachment probability. Clifford (1975) pointed out that corona from earthed and from isolated models differs greatly in character and in distribution A study of the correlation between corona distribution,strike distribution on models and strike distribution on full-size aircraft would be valuable.

6. CONCLUSIONS

The physical arrangement and dimensions for model tests are fairly well agreed. Provided that a large lower gap is present all authorities would be satisfied with isolated, suspended models of scale between 1:20 and 1:72, having a conducting surface and made to an accuracy of ± 0.5 mm. An upper gap of twice the model dimensions and a lower gap slightly larger than this would be regarded as satisfactory (or even luxurious) by any worker in the field. Smaller gaps would be acceptable to most.

Streamering tests would be accepted as indicative of possible attachment regions if the model is isolated. Strike point attachment tests would be needed to confirm the results.

Photography of the sparks from two directions, preferably at right angles, and inspection of the model surface are accepted techniques for determining attachment points. Normal high-voltage techniques for voltage and current measurements are also acceptable, and no diagnostic problems have required discussion.

The subjects causing disagreement among the experts are the pulse voltage wave shape, its polarity and its amplitude. Theoretical justification of any preference is difficult, and comparison with flight data for aircraft of different size and shape is the only convincing argument.

ACKNOWLEDGEMENTS

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QUESTIONS and ANSWERS

From G. Dubro

Q – I should like to make two brief comments. First, I would make comment on Dr Little's reference to Rühling observed rise time data of 125 msec. I believe this data comes from triggered lightning experiments with rockets. Indeed there is question whether such triggered lightning characteristics are adequately similar to natural lightning events, and hence justification for its applicability to the long wave front tests for attachment is really inconsequential. Secondly, this test has some controversy as related to test set-up and operating parameters, i.e. gap spacing, rise-time, electrode configuration, etc. It seems to me that a sensitivity analysis of these parameters would greatly aid the engineer in knowing what are the important parameters to control and the impact on test results by varying one of those parameters.

A – With regard to the first comment, the work of Rühling referred to deals with the protection of power lines and ground installations, so that the conclusions he draws apply to natural lightning strokes. They support the use of long-fronted waves in model aircraft tests.

I agree whole-heartedly with the second comment. More investigations of the test techniques to define the controlling parameters are greatly needed.

BIOGRAPHICAL NOTE

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He obtained his Doctorate at Oxford University, England, and spent five years as a research officer there afterwards. Since then he has worked for the UK Atomic Energy Authority at Harwell and Oxford, apart from some visiting professorships in USA. His research interests were originally in the field of plasma physics, but for the last four years he has worked on lightning protection for aircraft.

NATO UNCLASSIFIED

LABORATORY TESTS TO DETERMINE LIGHTNING ATTACHMENT POINTS WITH SMALL AIRCRAFT MODELS (ENGINEERING TEST)

by

D. W. Clifford

ENGINEERING LABORATORIES
MCDONNELL AIRCRAFT COMPANY
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MCDONNELL DOUGLAS CORPORATION

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by
Don W. Clifford
McDonnell Aircraft Company
St. Louis, Missouri, 63166

SUMMARY

High voltage laboratory generators exist which are capable of producing long sparks up to several meters in length. These long sparks are often used in conjunction with scale model aircraft to predict the location of lightning attachment points on new aircraft designs. The distribution of attachment points recorded on the model is used to define lightning attachment zones on the actual aircraft. These zonal definitions are used to determine the lightning threat for aircraft components and structures based upon their location on the aircraft.

This paper discusses the equipment and procedures used to conduct model attachment point studies. It further evaluates the effects of certain testing variables on the results of the test. It will be seen that although uncertainties exist in the proper definition of some test parameters, adequate agreement has been observed between the results of laboratory tests and actual lightning strike histories of numerous aircraft to give confidence to the test. On that basis, a set of test parameters is defined for conducting engineering R&D tests on scale model aircraft for lightning attachment point determinations.

1. INTRODUCTION

INTRODUCTION. The initial laboratory test used in the development of lightning protection for an aircraft is often a scale model lightning attach point study. It is important for aircraft designers to know precisely where lightning may strike on a new aircraft. It is known that lightning initially attaches primarily to the extremities of the aircraft. Most attachments are to wing tips, nose, horizontal and vertical fin tips and other protuberances such as antennas, probes or external stores. However, there are often questionable areas on most aircraft where a simple examination is not sufficient to determine, with any confidence, whether lightning will strike there or not. If there is a question of attachment probability, then it is likely that the probability of an attachment there is low. However, where safety of flight is concerned, even low probabilities must be considered. In such cases, more information is needed for making important design decisions regarding lightning protection. The placement of fuel vents, the incorporation of protective measures for composite surfaces, and the routing and shielding of critical avionics wiring may be influenced by the location of attachment points.

A scale model lightning attach point study can assist in this determination. The study is typically performed by placing a conductive model of the aircraft in an air gap between electrodes of a high voltage impulse generator. The air gap is overvolted, causing a long spark to propagate between the electrodes. The spark is intercepted by the aircraft model, and the points on the model where the spark attaches are recorded as possible lightning attachment points. The model is rotated in small steps through all three axes of roll, pitch and yaw, and spark discharges to the model are triggered in each position. In this manner, lightning strikes to the aircraft from every direction are simulated.

Typically, three to ten shots are taken with the model in each orientation. Photographs, usually taken with two cameras positioned at right angles to each other, are used to identify and record the attachment points. Data from the photographs can then be used to tabulate all attachment points or to construct diagrams of various types showing capture angles or probability distributions.

In a laboratory attachment point study, the beginning assumption is that the aircraft will be struck by lightning, and that it will be struck many times. This assumption is based on the fact that the model represents not one, but a whole fleet of aircraft flying for a period of years in a full range of weather conditions. Therefore, whereas one particular aircraft in the fleet may receive no more than a few strikes over its entire lifetime, when all of the strikes to all of the aircraft of that type in service are considered collectively, a statistically large distribution of strikes to different points of that aircraft will eventually be accumulated.

LIGHTNING ZONES. Attachment point tests identify directly the regions on an aircraft where lightning may initially attach. These regions are designated as Zone 1 (surfaces of the vehicle for which there is a high probability of initial lightning flash attachment). Since in actual flight the lightning arc channel may persist for large fractions of a second, an aircraft struck by lightning may fly through the stationary arc channel. This relative movement can result in an arc attachment point moving from an initial forward attachment, back over the surface of an aircraft, thus exposing aft in-board sections of the aircraft to lightning currents. The regions of the aircraft over which the arc may sweep in this manner are designated as Zone 2 (surfaces of the vehicle across which there is a high probability of a lightning flash being swept by the airflow from a Zone 1 point of initial flash attachment). Zone 3 includes all of the vehicle areas other than Zone 1 or Zone 2 regions. Zone 2 and 3 regions are inferred from the Zone 1 regions identified in the model attachment point test.

TEST PHILOSOPHY. In planning a model attach point test, a number of questions arise concerning various test parameters such as model size, construction and accuracy, air gap spacing, electrode geometry, electrode polarity, high voltage waveform, grounding of the model and the number of discharges at each model orientation. It is difficult to specify these variables exactly, because a detailed knowledge of the mechanisms of propagation of natural lightning and the associated fields does not exist. Furthermore, there is inadequate knowledge about the propagation mechanisms of simulated lightning generated in the laboratory and, therefore, the degree of correlation between laboratory and natural lightning.

Some disagreement exists, therefore, among lightning experts and aircraft designers as to the usefulness of laboratory spark attachment tests on models. Some workers feel that the lack of a rigorous theoretical understanding of all the mechanisms involved in lightning and laboratory sparks essentially invalidates the technique, except as a subject of further research. At the other extreme, since good agreement seems to exist between in-flight strike records and laboratory test results, the test is used by some workers as the primary basis for design decisions. These workers feel that if spark attachments to specific regions of a model aircraft cannot be produced in the laboratory test, those regions on the actual aircraft can be considered to be safe from lightning strikes. Critical safety-of-flight equipment may then be placed in those regions with no provision for protection, even though the equipment may be known to be vulnerable if struck.

Between these extremes of opinion, most workers feel that the test technique has value and is helpful in giving a general feel for the probability of strikes to questionable areas. The issue then becomes the degree of conservatism which should be designed into the test through the adjustment of certain test conditions which can affect the test results. As additional information is gained through continuing lightning and high voltage research, it is anticipated that the proper values of the test parameters will become better defined and that confidence in the laboratory test will increase.

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This paper discusses the equipment and procedures used to conduct model attachment point studies. It further evaluates the effects of certain testing variables on the results of the test. It will be seen that although uncertainties exist in the proper definition of some test parameters, adequate agreement has been observed between the results of laboratory tests and actual lightning strike histories of numerous aircraft to give confidence to the test. On that basis, a set of test parameters is defined for conducting engineering R&D tests on scale model aircraft for lightning attachment point determinations.

1. INTRODUCTION

INTRODUCTION. The initial laboratory test used in the development of lightning protection for an aircraft is often a scale model lightning attach point study. It is important for aircraft designers to know precisely where lightning may strike on a new aircraft. It is known that lightning initially attaches primarily to the extremities of the aircraft. Most attachments are to wing tips, nose, horizontal and vertical fin tips and other protuberances such as antennas, probes or external stores. However, there are often questionable areas on most aircraft where a simple examination is not sufficient to determine, with any confidence, whether lightning will strike there or not. If there is a question of attachment probability, then it is likely that the probability of an attachment there is low. However, where safety of flight is concerned, even low probabilities must be considered. In such cases, more information is needed for making important design decisions regarding lightning protection. The placement of fuel vents, the incorporation of protective measures for composite surfaces, and the routing and shielding of critical avionics wiring may be influenced by the location of attachment points.

A scale model lightning attach point study can assist in this determination. The study is typically performed by placing a conductive model of the aircraft in an air gap between electrodes of a high voltage impulse generator. The air gap is overvolted, causing a long spark to propagate between the electrodes. The spark is intercepted by the aircraft model, and the points on the model where the spark attaches are recorded as possible lightning attachment points. The model is rotated in small steps through all three axes of roll, pitch and yaw, and spark discharges to the model are triggered in each position. In this manner, lightning strikes to the aircraft from every direction are simulated.

Typically, three to ten shots are taken with the model in each orientation. Photographs, usually taken with two cameras positioned at right angles to each other, are used to identify and record the attachment points. Data from the photographs can then be used to tabulate all attachment points or to construct diagrams of various types showing capture angles or probability distributions.

In a laboratory attachment point study, the beginning assumption is that the aircraft will be struck by lightning, and that it will be struck many times. This assumption is based on the fact that the model represents not one, but a whole fleet of aircraft flying for a period of years in a full range of weather conditions. Therefore, whereas one particular aircraft in the fleet may receive no more than a few strikes over its entire lifetime, when all of the strikes to all of the aircraft of that type in service are considered collectively, a statistically large distribution of strikes to different points of that aircraft will eventually be accumulated.

LIGHTNING ZONES. Attachment point tests identify directly the regions on an aircraft where lightning may initially attach. These regions are designated as Zone 1 (surfaces of the vehicle for which there is a high probability of initial lightning flash attachment). Since in actual flight the lightning arc channel may persist for large fractions of a second, an aircraft struck by lightning may fly through the stationary arc channel. This relative movement can result in an arc attachment point moving from an initial forward attachment, back over the surface of an aircraft, thus exposing aft in-board sections of the aircraft to lightning currents. The regions of the aircraft over which the arc may sweep in this manner are designated as Zone 2 (surfaces of the vehicle across which there is a high probability of a lightning flash being swept by the airflow from a Zone 1 point of initial flash attachment). Zone 3 includes all of the vehicle areas other than Zone 1 or Zone 2 regions. Zone 2 and 3 regions are inferred from the Zone 1 regions identified in the model attachment point test.

TEST PHILOSOPHY. In planning a model attach point test, a number of questions arise concerning various test parameters such as model size, construction and accuracy, air gap spacing, electrode geometry, electrode polarity, high voltage waveform, grounding of the model and the number of discharges at each model orientation. It is difficult to specify these variables exactly, because a detailed knowledge of the mechanisms of propagation of natural lightning and the associated fields does not exist. Furthermore, there is inadequate knowledge about the propagation mechanisms of simulated lightning generated in the laboratory and, therefore, the degree of correlation between laboratory and natural lightning.

Some disagreement exists, therefore, among lightning experts and aircraft designers as to the usefulness of laboratory spark attachment tests on models. Some workers feel that the lack of a rigorous theoretical understanding of all the mechanisms involved in lightning and laboratory sparks essentially invalidates the technique, except as a subject of further research. At the other extreme, since good agreement seems to exist between in-flight strike records and laboratory test results, the test is used by some workers as the primary basis for design decisions. These workers feel that if spark attachments to specific regions of a model aircraft cannot be produced in the laboratory test, those regions on the actual aircraft can be considered to be safe from lightning strikes. Critical safety-of-flight equipment may then be placed in those regions with no provision for protection, even though the equipment may be known to be vulnerable if struck.

Between these extremes of opinion, most workers feel that the test technique has value and is helpful in giving a general feel for the probability of strikes to questionable areas. The issue then becomes the degree of conservatism which should be designed into the test through the adjustment of certain test conditions which can affect the test results. As additional information is gained through continuing lightning and high voltage research, it is anticipated that the proper values of the test parameters will become better defined and that confidence in the laboratory test will increase.

EXPERIMENTAL VARIABLES. In view of the fact that model attach point tests may be relied upon to make important lightning protection design decisions, the test should be conservative, but as realistic as possible. The large number of laboratory variables which must be considered, along with the uncertainties already mentioned, makes the establishment of a standard test procedure difficult. However, agreement has been reached on most aspects of the technique, with optional procedures defined in the principal areas of uncertainty.

Three categories of variables will be considered. They are:

a. the high voltage waveform,

b. the test model, and

c. the test setup and procedure

THE HIGH VOLTAGE WAVEFORM. The questions surrounding the proper choice of high voltage waveshape and polarity are probably the most complex, and at the same time the most significant in terms of the effect of the choice on the results of the test. It has been shown that the use of high voltage pulses with fast wavefronts, such as the standard 1.5/40 wave (1.5 μsec rise, 40 μsec decay), results in a limited distribution of attach points, mostly in high field regions. However, if slow rising waves are used, such as a 200 μsec rise time switching pulse, then the arc propagation mechanisms are altered, and the arc tends to meander in an erratic fashion. As a result, the arc attachment points become more scattered, resulting in a larger number of attachment points, including some at inboard regions. Since the model attachment points produced by the slow wave are more scattered, the test is seen as being more conservative, thus producing greater confidence that all attach points have been considered.

The latter point is certainly well taken, as long as the results are realistic. However, it may be possible, by adjusting the test parameters, to force laboratory arcs to attach to points where natural lightning might never attach. Design decisions based on such a test could result in overdesigned protection or unnecessary and costly design compromises. Any unnecessary weight, cost or performance penalties, particularly for fighter aircraft, must be avoided. Therefore, realism in the attachment point test is essential.

Fast rising waveforms have been used in most of the previous model tests, resulting in fewer attach points but good correlation with flight experience. Extensive flight data are available recording actual lightning strikes to aircraft, but no strike attachments have been reported to areas of aircraft which apparently would not be predicted by fast rise time simulation tests. Many instances of strikes on the fuselage, aft of forward attachments, have been reported; these attachments are probably swept strokes. However, no evidence of direct strikes to inboard wing sections or similar low field regions have been reported.

As a result of the waveform controversy, a double-option approach has been agreed upon by workers in the U.S. and Britain. Two test waveforms have been defined for model attach point testing. The first is a fast-fronted waveform which is to be used for what is termed "Fast Front Model Tests." The second waveform is a slow rising waveform which is to be used for "Slow Front Model Tests."

The voltage waveform for the fast front model test is a chopped voltage waveform in which flashover of the gap between the model under test and the test electrodes occurs at two μsec (±50%). The amplitude of the voltage at the time of flashover and the rate-of-rise of voltage prior to breakdown are not specified. Attachment points on models obtained by tests using this waveform are all automatically designated as Zone 1 regions.

The voltage waveform used for the slow front model tests has a rise time between 50 and 250 μsec so as to produce a more erratic arc attachment behavior. It should produce a greater spread of attach points, possibly including attachments to low field regions. Therefore, the test data obtained must be analyzed by appropriate statistical methods to arrive at the definition of *Zone 1 regions. In other words,* some judgment must be exercised in interpreting the results of the slow front test, whereas the fast front test results are used directly. Neither criteria for judgment, nor the "appropriate statistical methods" to be employed in the slow front test analysis, have been specified.

THE TEST MODEL. The definition of the model must include considerations of model size, model accuracy, and model construction. The question of model size has generally been addressed on the basis of the maximum size which can be accommodated by the test facility. Since the air gaps required between the model and the high voltage electrodes are generally taken as some multiple of the model size, the upper limiting factor is the length of arc which can be generated. Arc length is a function of generator voltage, capacity and impedance. Impulse generators used for attach point testing vary in size from about one to five megavolts, corresponding to arc lengths from about two to seven meters. In order to maintain a total gap to model ratio of at least 2.5:1, model sizes could not exceed 0.5 meters for the smaller facilities. However, the larger facilities could accommodate models up to 2.0 meters in length.

It is difficult to define quantitatively the minimum model size acceptable. Generally, economic considerations will require that the model size be limited to the least expensive size acceptable. It has been shown that the smaller the model, the more important the accuracy of the model becomes. Surface irregularities or other imperfections may become a significant factor on very small models, whereas on larger models, they would be insignificant.

Tests have been conducted[1] on various size models of fighter aircraft, using a 1.5 Mv fast wave impulse generator. Essentially identical results were obtained on 5%, 3%, and 1.4% models, although effects of flaws in the surface finish of the 1.4% model were noted.

Based on these considerations, generally the scaling should be restricted to no lower than three percent, although for large vehicles, it may be reasonable to scale down to one-to-two percent without surface irregularities becoming a factor. On the upper limit, facility limitations will generally restrict model scales to ten percent or lower.

MODEL ACCURACY. The second model factor, accuracy, was also addressed in the test series involving the three different size models discussed above. The five-percent model was an extremely accurate wind tunnel model with dimensions held to ±0.024 mm. The three-percent model, on the other hand, was purchased commercially as an accurate plastic model. It was painted with a conductive paint and then electroplated with a 0.13 mm nickel coating. An extensive attach point mapping was conducted on both models with identical results. The accuracy of the plastic model after plating was estimated to be about ±0.5 mm on the mold lines. Some inboard surface detail was not reproduced on the plastic model, but no arc attachment was observed to those details on the accurate model. It was, therefore, decided that extreme model accuracy is not necessary, although the tolerance on mold line accuracy becomes more critical on smaller models. In the referenced tests, it was determined that a ±0.5 mm tolerance is acceptable on a model 0.3 meters long.

MODEL CONSTRUCTION. The question of model construction arises from both economic and technical considerations. For example, the intricate solid metal models built for wind tunnel aerodynamic studies can be used for model testing if they are available. Otherwise, they are unnecessarily expensive and usually too heavy for convenient attach point testing. Sufficiently accurate models can be constructed fairly inexpensively by comparison. Since the high voltage arcs used for attach point testing are relatively low current and low in energy content, only a thin metallized coating of a few tenths of a millimeter is required on nonmetallic models. Care should be taken to ensure that large nonconducting surfaces on the aircraft are also nonconducting on the model.

THE TEST SETUP AND PROCEDURE. Within the category of test setup and procedure, the variables of electrode geometry, polarity, air gap spacing and model position in the gap, and number of discharges will be considered.

ELECTRODE GEOMETRY. Although both rod-to-rod and rod-to-plane electrode geometries have been used for model studies, a rod/plane geometry is generally chosen with the rod representing the oncoming stepped leader and the plane representing the opposing diffuse charge center. However, if an adequate spacing is maintained between the ground electrode and the model, the irregular field distribution for the two geometries would likely be indistinguishable at the model.

AIR GAP SPACING. The question of air gap spacing between model and electrode has been addressed empirically by Stahmann[2] and somewhat more analytically be James and Philpott[3]. Stahmann concludes that the gap spacing should be the minimum gap for which no variation of attach point behavior occurs with further spacing. This approach logically removes gap spacing as a variable, unless other scaling considerations indicate that the gap spacing should be less than this minimum empirical value. In practice, Stahmann's minimum gap works out to be about 1.5 model dimensions, which translates to about one to three meters for the range of model sizes generally used. Using calculated field strengths and variations in field strength with distance of the stepped leader from the aircraft, James and Philpott confirm that a primary air gap to model dimension ratio of 1.5 to one can be used. The calculated stress does not rise significantly for distances between leader and aircraft greater than 50 meters. Using the scaling factors quoted for the aircraft model, the 50-meter value would scale down to a few meters for the test gap.

Consideration of model to ground plane spacing has received less attention, with the distance nominally set at one model length as a safe value. This setting is most likely deduced from mirror image considerations in the ground plane. However, in addressing the question of grounded versus ungrounded models, field measurements and calculations by Shaeffer[4] indicate that the model can be located somewhat closer to the ground plane without affecting field distributions between the model and rod electrode.

On the other hand, high voltage arc attachment experiments to simplified geometry test articles mounted on an insulated test stand were reported by Philpott, et.al.[5] In those experiments, variations in the model to ground spacing up to 2.5 model dimensions resulted in changes to the attachment distribution. Consequently, some workers prefer longer model to ground gaps. For standard testing, the model to ground spacing is specified to be in the range of one to 2.5 model dimensions.

NUMBER OF SHOTS. Both Stahmann[2] and James and Philpott[3] have attempted to define the number of strikes from each position in order to ascertain low probability levels. The approaches differ fundamentally, in that Stahmann assumes that ten strikes from a given electrode position will adequately identify the possible attach points from that position. Probabilities of lightning attachment to each point are then determined by the ratio of the solid angle through which strikes may come to attach to that point, to the total 4π angles possible. In other words, a sphere surface is defined with the model at the center and with a radius defined by the high voltage electrode spacing. Then the probability of lightning striking a designated point on the aircraft is the ratio of the surface area on the sphere defined by electrode positions which will allow strikes to the point, to the total surface area of the sphere. With this approach, Stahmann used only a total of 30 shots to establish a 10^{-4} probability of strike to a given model region.

By this definition, if no electrode position is found from which arcs will strike a designated point (or area), using no more than ten arcs per position, then the zone is considered to be safe with negligible probability of lightning strikes. The number ten is arbitrary, but is a reasonable limit for a test program where numerous electrode positions may be required for a full vehicle mapping. James and Philpott have suggested that 100 shots at each position may be necessary to establish less than a one percent strike probability. For Engineering Test purposes, the specification calls for three to ten shots at each position.

ELECTRODE POLARITY. The choice of electrode polarity can influence the results of the attachment point test. A positive high voltage electrode will produce a more erratic arc propagation and, therefore, a greater distribution of attachments. In actual lightning conditions, it might be argued that since the stepped leader (at least in cloud-to-ground discharges) carries negative charge in the great majority of cases, then the laboratory H.V. electrode should be of negative polarity. However, the question of the degree of conservatism desired again enters into the consideration. It is generally accepted that the influence of electrode polarity is smaller than the influence of the high voltage waveform, and is, therefore, of only secondary concern. Perhaps, in order to add a degree of conservatism to the fast front test, a positive electrode polarity has been accepted as the standard polarity for both tests.

SUMMARY. Model attach point tests are often used to predict lightning attach points and swept stroke lightning zones on new aircraft. These tests are used to determine attach point distribution probabilities and to determine safe or shielded areas where vulnerable systems may be located. For non-safety-of-flight equipment, strike point testing can define economic tradeoffs where maintenance and repair costs can be weighed against the problems of incorporating additional protection.

Concerns over test variables affecting the results primarily revolve about the high voltage waveform. Some investigators prefer the use of slow waves with rise-times on the order of hundreds of microseconds, coupled with positive polarity high voltage electrodes. Each of these is calculated to produce a more conservative test because of the wide scatter of attachment points produced. However, since no lightning strikes to aircraft in flight have been reported to regions of aircraft other than those which might be indicated by fast wave model tests, either fast or slow front waveform tests may be used. The slow waveform test results must be interpreted carefully in the determination of Zone 1 regions on the aircraft.

Extreme model accuracy is not critical as long as the model is large enough to mask small imperfections in the finish. Either rod-to-rod or rod-to-plane electrode geometries can be used if the model is removed from the ground electrode by one model length. Spacing between the model and H.V. electrode is taken to be 1.5 model lengths, both from empirical and analytical scaling considerations.

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QUESTIONS and ANSWERS

1 – From J. Taillet

Q – Have you observed, in the model tests, any strikes in absence of streamers from the aircraft model ?

A – No, we have not. However, in the normal procedure of conducting arc attachment tests, the intensity of the arc channel precludes the observation of preionization events, such as streamering, on the photographic records. In situations where we have photographed streamering activity from a model, and then have separately conducted arc attachment tests, the arcs have always attached to points on the model where streamering activity was observed to be the most intense. Such studies are rather few in number, however, and a more extensive investigation would be required to state conclusively, on the basis of experimental evidence, that arcs never attach to a model in the absence of streamers. However, the laboratory evidence that we do have is consistent with our understanding of the process of natural lightning attachment to aircraft in flight. That is, the intense electric field produced by the oncoming stepped leader causes localized air breakdown (corona) at the highest field regions on the aircraft. As the field continues to increase, streamers propagate towards the oncoming leader, balanced electrically on the opposite side of the aircraft, by streamers of the opposite polarity propagating towards ground or the other charge center. When the stepped leader contacts the streamer from the aircraft, both streamers and the aircraft become an extension of the stepped leader so that when the arc channel path is complete, the return stroke current will flow through the points on the aircraft where the primary streamers initiated. Those points are what we call arc attachment points.

2 – From A.W. Bright

Q – I would imagine that areas of intense ionization, such as static discharger points, would have a considerable influence on the location of lightning attachment points on large aircraft. When making model tests on large aircraft, is any attempt made to simulate the influence of static dischargers ?

A – To my knowledge, no one has attempted to account for the influence of static dischargers in a model test. One reason is that it would be difficult to scale an 8-inch long, 1/4-inch diameter discharger on a 1/72 scale model. It would be imperceptibly small. Another reason is that the static discharger points are resistively decoupled from the aircraft structure so that the charge available to feed streamers is limited by the series resistance of the discharger. However, if the field at the discharger is high enough, a streamer from the main structure may flash over the resistive shank and lightning may strike the discharger. However, flight experience has shown that strikes to static dischargers are not frequent, so it is assumed that their influence on the arc attachment process is not a major concern.

BIOGRAPHICAL NOTE

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Mr. Clifford supervises the Engineering Physics Laboratories at McDonnell Aircraft which include the Applied Optics and Laser Effects Lab., the Thermophysical Properties Lab., the Ion Vapor Deposition Facility and the Lightning and High-Voltage Laboratories. He received his BS in Engineering Physics from the University of Kansas in 1962 after four years in the U.S. Navy as a Gunfire Control Technician. He began his professional career as a Research Engineer for Atomics International, where he operated an experimental facility engaged in the development of compact nuclear reactors for space applications.

After joining McDonnell in 1965, Mr. Clifford earned an MS degree in Engineering Management from the University of Missouri. At McDonnell he has been responsible for the development of numerous experimental facilities involving high-energy electrical discharge apparatus. These facilities include a 100-kilojoule Magnetically Accelerated Plate-Slap system for nuclear weapons effects studies, multimegavolt impulse generators and extensive high-current lightning simulators, including a 1.1 megajoule system which is the largest known high-current lightning simulator. Mr. Clifford is a member of Sigma Pi Sigma Physics and Tau Beta Pi Engineering Honor Societies, and the Institute of Environmental Sciences. He is a recipient of the AIAA Young Professional Award.

NATO UNCLASSIFIED

LABORATORY SIMULATION OF SWEPT LIGHTNING STROKES

by

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MCDONNELL DOUGLAS
CORPORATION

NATO UNCLASSIFIED

LABORATORY SIMULATION OF SWEPT LIGHTNING STROKES
(ENGINEERING TEST)

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SUMMARY

When an aircraft is struck by lightning in flight, the aircraft may fly through the stationary arc channel which often persists for large fractions of a second. Consequently, the lightning attachment point appears to traverse the surface of the aircraft, carried by the windstream. This dynamic sweeping effect results in lightning attachment to surfaces of the aircraft which would not be struck directly The nature of the sweeping action and the damage produced by a swept stroke can be simulated by testing. Various techniques have been employed, to either move the test surface through a stationary arc or to blow the ionized arc channel over the test surface. This paper describes the swept stroke phenomenon and important test parameters in swept stroke simulation. The importance of test article configuration, air flow, electrical current characteristics, and the behavior of high-current arcs is discussed.

1. INTRODUCTION

1.1 THE MOVEMENT OF AN AIRCRAFT THROUGH A LIGHTNING STRIKE. Figure 1 summarizes the work of Cianos and Pierce,[1] which shows the essential parameters of a typical lightning strike as a function of time. In this model, an initial peak current spike up to 200 kamps lasting 100 μsec is followed at intervals by lower level restrikes of the same general shape, i.e., very fast rise, slow decay. Some strikes may die out very slowly – these long surges carry heavy currents and a large transfer of charge. Since the current doesn't die out completely between restrikes, the arc channel remains ionized so that restrikes follow the same path as the initial strike. Figure 2 shows a simplified lightning current waveform having all the strike components condensed into a single composite waveform. The initial high-peak current pulse is followed by intermediate and continuing current surges, and the stroke is terminated by a high-peak current restrike.

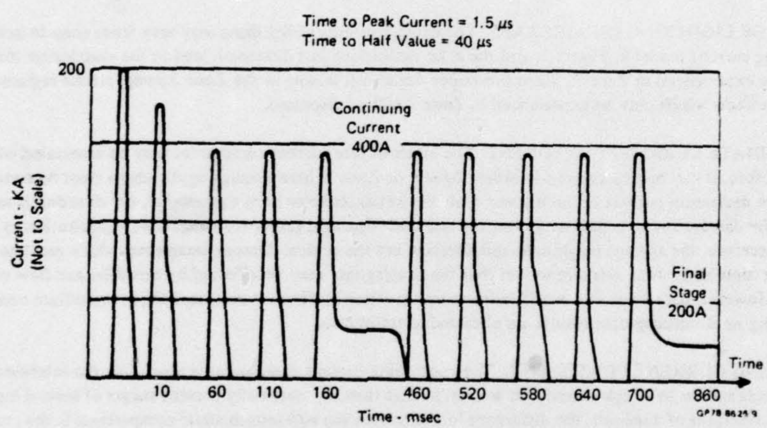

Figure 1 - Model of Natural Lightning

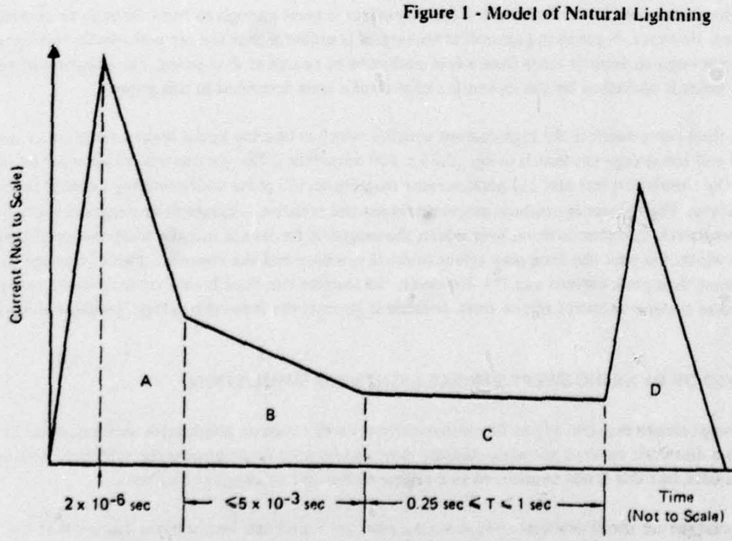

Figure 2 - Idealized Current Test Waveforms for Evaluation of Direct Effects

1.2 LIGHTNING ATTACH POINT ZONES ON AIRCRAFT. The study of dynamic interactions of lightning arc channels with aircraft in flight has led to a definition of aircraft zones based on the configuration of the aircraft.[2] Zone 1 is defined as the high-electric field regions around the extremities where lightning can initially attach. Since the initial high-current spike lasts only microseconds, it will affect only the direct attach point. Zone 2 is defined as the surfaces aft of the forward direct attach points where the decaying arc channel could sweep, and thus expose these regions to lightning currents occurring after the initial high-current spike. Zone 2 is commonly referred to as the Swept Stroke Zone, and is where many potentially vulnerable safety-of-flight structures are located. Zone 3 is defined as those sections of the aircraft where lightning will not attach at all. However, lightning currents may pass through Zone 3 structures enroute through the vehicle. The three zones are determined variously by simple inspection, analytical or laboratory techniques which generate electrostatic field distributions, or finally by lightning attach point tests on models.[3,4,5] Figure 3 illustrates the zonal regions on a commercial aircraft.

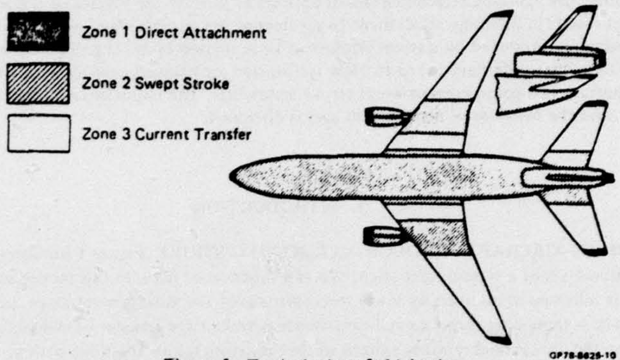

Figure 3 - Typical Aircraft Lightning Strike Zones

1.3 EFFECTS OF LIGHTNING ON AIRCRAFT. The damage effects of lightning may vary from zone to zone on an aircraft. The simplified lightning current model in Figure 2, and the zone definitions just discussed, lead to the conclusion that only current components B, C, and D will be experienced in Zone 2. Since this paper deals with testing in the Zone 2 swept stroke regions, lightning strike components and related effects which may be experienced in Zone 2 will be discussed.

1.4 INTERMEDIATE CURRENT COMPONENT. The intermediate current component may be associated with either an initial strike or a restrike; therefore, it can be experienced in either Zone 1 or Zone 2. Many design applications must be concerned with this component. Although the maximum current is much lower than the initial strike or even the restrike, the duration is much longer, being measured in milliseconds. The damage is characterized by heating and burn-through, rather than explosive vaporization as in the case of high-peak current spikes. Therefore, the current magnitude and duration are the critical damage parameters which must be closely simulated. The five-msec time duration leads some workers to feel that the heating rate may be affected by aerodynamic flow over the surface, producing a cooling action. However, data from sled tests[6] indicate no significant differences exists. Still, intermediate current testing may be conducted with moving air if unacceptable results are obtained in static tests.

1.5 CONTINUING CURRENT COMPONENT. The continuing current component is much like the intermediate current component. Both produce effects similar to a high-current arc welder, in that they are essentially current surges of several hundreds of amps which may continue for large fractions of a second; the difference in the continuing and intermediate components is the current amplitudes and the time intervals over which they are applied. The continuing current component is very damaging when allowed to concentrate all its energy at one point. A few hundred coulombs of charge may be transferred and the heating effect is great enough to burn through thick metal panels if the arc remains attached at one spot. However, because the aircraft is moving, it is unlikely that the arc will remain stationary at any one point in critical Zone 2 regions long enough to deposit more than a few coulombs of charge at that point. The question of how long the arc will remain attached at a single point is addressed by the dynamic swept stroke tests described in this paper.

1.6 HIGH-CURRENT RESTRIKE. The final component is the high-current restrike which is like the initial high-current strike, except that the magnitude is one-half (100 kamps) and the energy content is lower (0.25×10^6 amps^{2}sec). The parameters which must be produced and controlled in a high-current restrike simulation test are: (1) peak current magnitude, (2) pulse width (energy content referred to as I^2t), and (3) in some tests, current rate-of-rise. High currents produce magnetic forces and resistive (explosive) heating and voltage gradients which are direct functions of current level. The time interval over which the magnetic forces act and the total energy dissipated in the test article are functions of the pulse width. Current rise time may affect internal sparking and the concentration of damage in resistive materials, but is normally less significant than peak current and I^2t. However, the restrike rise time is very critical when evaluating nonconductive surfaces or dielectric protection systems in swept stroke tests, because it governs the inductive voltage gradient along the arc channel.

2. OBJECTIVES OF DYNAMIC SWEPT STROKE LIGHTNING SIMULATION

The primary objectives of the dynamic swept stroke test are: (1) to determine current dwell times on conductive surfaces, and (2) to determine restrike-produced reattachment on dielectric covered surfaces. Airflow may also be used to determine the effects of aerodynamic cooling on long duration burning damage results, but this is not considered as a primary objective of swept stroke tests.

2.1 DWELL TIME TESTS. Continuing current arc dwell times in a swept stroke zone are important because it is known that the total continuing current component can burn through normal thickness metal skins. However, if it can be shown by tests that under airflow conditions the arc does not dwell in one place long enough to cause significant damage at that point, then it can be argued that the system is safe.

The swept stroke dwell time test should be conducted using realistic continuing current levels in order to accurately reproduce arc attachment and arc damage effects. It is also necessary to realistically simulate the aerodynamic interaction of the aircraft surface with the arc channel.

2.2 DYNAMIC RESTRIKE TESTS. The restrike is only important in a swept stroke test when considering a dielectric surface. For conductive surfaces, it should be assumed that a restrike may occur anywhere in a swept stroke zone. This is true because the restrike will always generate enough voltage along the arc channel to cause a reattachment if the arc is distended at all over a conductive surface. Therefore, an actual swept stroke test is not necessary to determine restrike attachment on metal surfaces; the damage effects of a restrike arc can be evaluated with a static damage waveform test at the worst-case location on the test surface.

Based upon some laboratory investigations, it has been proposed that critical Zone 2 surfaces may be protected from lightning by covering them with a layer of insulating material. Good dielectric films bonded to the surface may prevent the arc channel from attaching to the protected surface. The arc would presumably jump across the protected surface to reattach on the other side. In other cases, nonconducting structural materials, such as fiberglass, may be employed in a Zone 2 region, beneath which critical components may be located.

For dielectric protection systems, or for unprotected nonmetallic structures, a restrike situation is the worst-case condition for puncture. The voltage drop along the arc channel is greatly increased when a restrike occurs. If the dielectric strength of the surface is not sufficient to withstand the restrike potential, the arc channel will puncture the dielectric coating and attach to conductive structure beneath.

The restrike test does not necessarily require an accurate simulation of the heavy continuing current component of a lightning strike; however, there must be enough continuing current flow to initiate the arc and sustain it until the restrike is triggered. Not every natural lightning stroke includes a heavy continuing current component, but most do have restrikes, even if there is no substantial continuing component.

In order to produce realistic restrike voltage gradients in the arc channel, a full magnitude 100 kA pulse rising to peak in two μsec or less should be used. The rate-of-rise of the voltage at the output of the restrike generator should be very rapid, at least 1000 kV per microsecond.

The restrike current waveform can be low in energy content because the objective of the test is not to determine damage, but rather to determine the possibility of dielectric puncture, which is a voltage phenomenon. When return stroke currents begin to flow, the voltage generated in the arc channel is the sum of the resistive and inductive contributions ($iR + L di/dt$). Therefore, the peak current, i, and the current rate of change, di/dt, must be accurately reproduced. The R and L factors will be simulated adequately if the aerodynamic aspects of the test are accurate.

In practice, it is not always possible to produce a full 100 kamp, two-μsec restrike waveform for swept stroke testing. Very large and expensive generators are required. However, for engineering purposes, lower current levels can be used and the results can be extrapolated to the full threat level.

By using a low-energy restrike waveform (low I^2t), many tests can be conducted on the same test article because the arc produces little damage to the test surface. The restrike pulse is timed to occur at preselected positions of the arc channel over the surface by using separate trigger pulses to initiate the continuing current and the restrike. The restrike trigger can be electronically timed to occur (usually 2 to 20 msec after arc initiation) when the knee of the arc channel is over the middle of the test area. The maximum span which can be protected with a given dielectric system is then determined by increasing the size of the area of the test surface until the restrike punctures the surface, rather than following the ionized channel back to the forward attach point. If reduced current levels or slower rate-of-rise restrikes are employed, the size (or the span) of the safe area must be extrapolated (reduced) accordingly.

3. APPROACHES TO GROUND-BASED SIMULATION OF SWEPT STROKES

A number of techniques have been employed in attempts to simulate the effects of lightning sweeping across aircraft surfaces. For the purpose of discussion, these techniques can be divided into the cagetories of: (1) moving test articles and (2) moving arcs.

3.1 MOVING TEST ARTICLES. It seems logical that the most realistic way to simulate swept strokes would be to move a test surface through an arc channel, as an aircraft moves through the stationary lightning channel. However, since full-scale surfaces are required to realistically measure dwell times or arc skip distances, and since actual aircraft velocities are also required, an immediate problem is encountered. Only high-speed sled test facilities offer a reasonable possibility for such a full-scale, high-velocity test. However, because of the expense and logistical complexity of this approach, only one such test series has been conducted. Reference 6 reports the results of a series of swept stroke tests conducted by Culham Laboratory in the U.K. at the Royal Aircraft Establishment (Farnborough). The sled was used to carry a simulated wing at speeds up to 160 mph through an established continuing current arc channel. Arc dwell times were measured on various surfaces and materials with different boundary layer conditions. No attempt was made to conduct restrike tests.

Another notable attempt at moving a test surface through an arc channel is reported in Reference 7. This test differed from the British sled test in two ways. First, much slower velocities were employed because the test article was mounted on an automobile whose velocity was limited by the confined test area. Velocities around 40 mph were attained. Secondly, the test was basically a restrike test rather than a dwell time test. The generator used was a high-voltage AC transformer which produced a rapid series of long sparks (up to eight feet), some of which followed the ionized channel produced by the preceding arc, and some which did not. The technique was used to evaluate dielectric structures such as fiberglass fuel tanks for swept stroke punctures.

Attempts have been made at moving small test surfaces through arcs in the laboratory by using rotary devices, but the results are difficult to interpret and to scale. These difficulties, coupled with the severe limitations of the approach, have resulted in very few attempts to utilize it.

3.2 MOVING ARC CHANNEL. Two approaches to moving arc channels have been employed. A magnetically-driven arc has been employed in the U.K.[6], whereas most of the swept stroke testing in the U.S. has employed wind tunnels to produce windblown arcs.[8,9] The

wind tunnel approach has obvious advantages in being able to simulate the aerodynamic conditions, whereas the magnetic technique requires no wind tunnel with its accompanying facility complications. Arc dwell times reported from the magnetic test have been reasonably consistent with values observed in flight and in windblown arc tests.

The other technique, which has been employed in the U.S., uses wind tunnels to move the arc over a stationary test surface. Wind tunnel design and aerodynamic model testing of aircraft are well developed technologies. Extensive correlation exists between wind tunnel test results and actual flight behavior. It can, therefore, be assumed that there are no inherent limitations in the accuracy of the aerodynamic simulation using properly designed wind tunnels. Essentially, only the frame of reference is reversed. The arc channel moves with the surrounding air and is, therefore, stationary with respect to it, which is a realistic condition. Therefore, a properly designed wind tunnel swept stroke test should be able to produce a fully realistic simulation of the interaction of a lightning arc with a moving aircraft. The details of the wind tunnel approach with examples from the facility at McDonnell Aircraft Company (MCAIR) in St. Louis are discussed in the following section.

4. DETAILS OF WINDBLOWN ARC TESTS

The arc channel sweeping motion and reattachment behavior on a moving aircraft will depend on several factors, including the aerodynamics of the flight vehicle, the material properties of the swept surface, and the dynamic electrical properties of the lightning arc. Aerodynamic properties of the flight vehicle include velocity of flight, angle-of-attack, and aerodynamic configuration of the surface in question. The material properties include type of material and surface finish, i.e., surface conductivity, whether the surface is smooth or rough, painted or unpainted. If the material is a dielectric, any conducting materials in the vicinity, such as wiring, plumbing, or metal edging, may also affect reattachment behavior. The dynamic electrical properties of the arc include the conductivity, current density, the extension of the arc from previous attachment location, the time of occurrence of restrikes, and the rate-of-rise of the restrike current. This large number of variable which must be controlled in order to have a realistic test requires close attention to detail in order to make the test meaningful.

On the basis of an evaluation at MCAIR, it was determined that the windblown arc with a high-current restrike is by far the best available simulation of a swept lightning stroke. It must be noted, however, that the windblown arc test can be invalidated by the improper design or control of several test elements. There are four basic elements to the windblown swept stroke test. They are:

(1) The test article
(2) The wind source
(3) The continuing current simulation
(4) The restrike simulation

A block diagram of a system comprising these elements is shown in Figure 4.

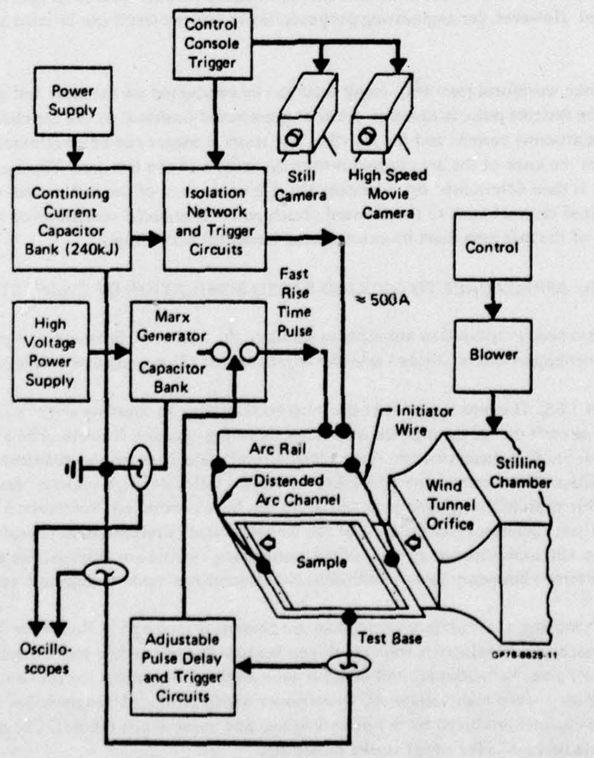

Figure 4 - Simplified Schematic of Swept Stroke Lightning Simulator

4.1 THE TEST ARTICLE. For simulation tests directed to the evaluation of a specific aircraft surface, surface characteristics of the test article should be a faithful representation of the corresponding aircraft properties. In particular, surface composition, finish, and irregularities can affect arc movement across the surface. The test article must be large enough to evaluate the arc behavior of interest. Dwell time

tests can be conducted on panels four feet or more in length along the direction of wind flow. The panels should be at least one foot wide. Multiple test runs should be planned because of the random nature of arc attachment behavior. Several tests can be conducted on the same test panel if it is two to three feet in width.

Restrike tests may require longer panels, six to eight feet in length. The magnitude and rate-of-rise of the restrike determine the amount of scaling required to relate the test results to the severe lightning case. The amount of scaling required may be a factor in determing the panel size.

Flat panels should be appropriate for worst-case tests since boundary layer thickness is not expected to play a major role in arc reattachment. For all practical purposes, air movement within boundary layers over critical surfaces will be turbulent. Therefore, arc channels distended within the boundary layer, regardless of its thickness, will be in virtual contact with the test surface at all points along its length. Consequently, a flat panel mounted flush with the floor of a laminar flow wind tunnel will accurately simulate arc attachment behavior for any practical angle-of-attack or air foil curvature.

4.2 THE WIND SOURCE. The wind source is extremely important in this type of test. The turbulence produced by rotating fans or blades must be eliminated in order to produce the smooth laminar flow environment which is essential to useful simulation. Any turbulence which would distort the arc channel and prevent it from lying smoothly in the thin boundary layer against the test surfac would nullify the test results since the objectives of the test are a direct function of the arc reattachment behavior. Turbulence can be reduced by the use of a stilling chamber through which the air can be expanded and then contracted again to produce the desired airflow conditions.

Screens and vanes can be added in the chamber and ducting to further reduce turbulence. At MCAIR, tests conducted with a centrifugal blower producing turbulent flow were compared with the results of tests conducted in the new swept stroke facility which produces very good laminar flow. Comparative still photos of the two conditions are shown in Figure 5. Dwell times are invariably longer in the turbulent case. An examination of the high-speed movies show that turbulence often causes the arc channel to twist about and close upon itself, forming a new current path upstream of the original arc channel. The photographic comparison also makes obvious the distortion of the arc channel in the turbulent case and clearly illustrates the necessity for laminar flow.

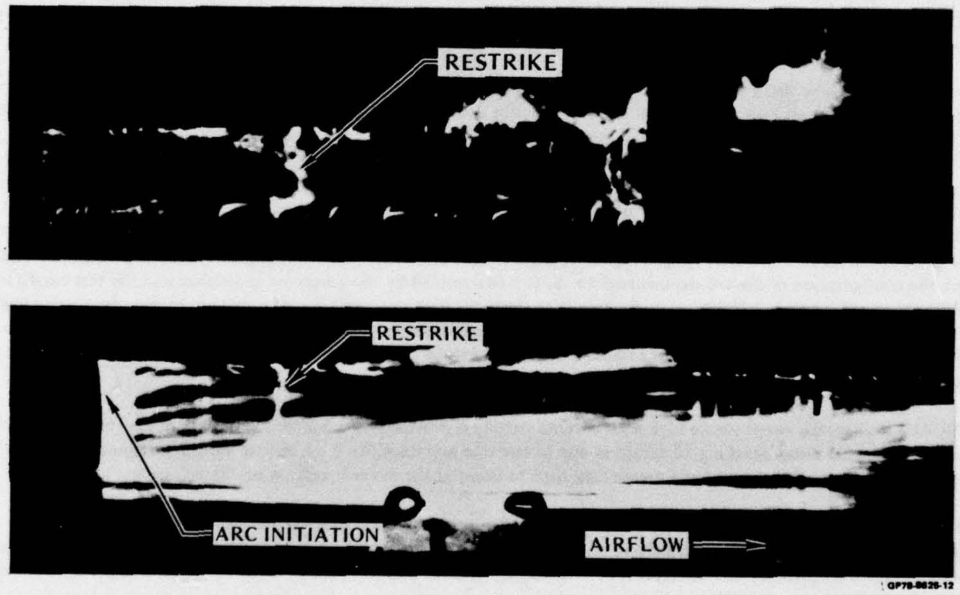

Figure 5 - Swept Stroke Lightning Test (Top) Turbulent Air Flow (Bottom) Laminar Air Flow

The velocity of the windstream should be representative of landing speeds of the aircraft under study. Too fast a velocity will produce shorter dwell times and not be representative of worst-case conditions. Too slow a velocity will produce overly severe results and could result in an overtest.

The MDC swept stroke air supply is generated by an Allis Chalmers Model 1007 compressor which produces a mass flow of 65 lb/sec, driven by a 3000-hp electric motor. The air is expanded into a screened 400-cubic foot stilling chamber, and then contracted by a ratio of 18:1 to exit into the atmosphere through a 12 x 30-inch rectangular nozzle. The large contraction ratio results in a windstream that exhibits very good laminar flow characteristics over the length of the test article. Conventional low-speed wind tunnels require a ratio of at least 7:1. However, for true laminar flow, a ratio of at least 16:1 is required.

The windstream velocity at the nozzle of the MCAIR tunnel is continuously adjustable up to 250 knots and is stable at a given velocity to within 0.5 knots. The velocity profile across the windstream has been mapped at various distances downstream from the nozzle, out to 7.5 feet. It can be seen from the curves in Figures 6 and 7 that the centerline velocity degrades less than six percent over an eight-foot long test section, and the windstream nonuniformity [illegible] percent.

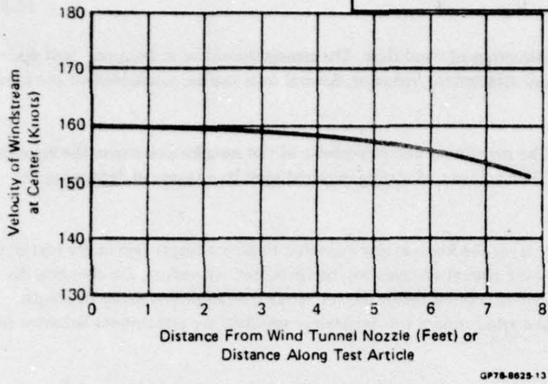

Figure 6 - Typical Velocity Profile of the Windstream

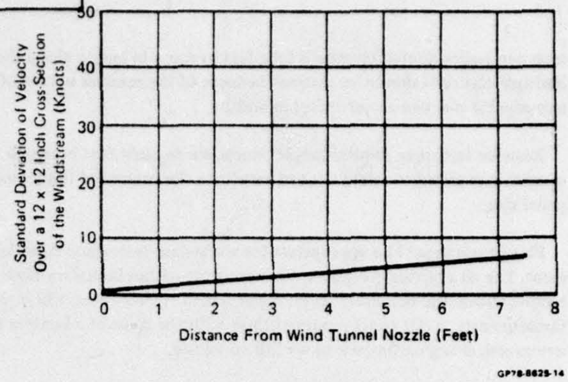

Figure 7 - Typical Cross-Sectional Velocity Profile of the Windstream

5. HIGH CURRENT GENERATORS

5.1 THE CONTINUING CURRENT SIMULATION. The continuing current arc simulation is a fairly straightforward, since all that is required is a reasonably constant DC current of a few hundred amps with enough voltage to sustain the arc, even though it may be stretched out to several feet in length. The latter requirement generally rules out batteries or a DC generator, unless a large inductive clamp is employed, since the voltage required is thousands of volts and the resulting steady-state wattage requirements are prohibitive. A large capacitor bank can store sufficient charge to produce an RC decay lasting over several tens of milliseconds with a driving voltage of several thousands of volts. However, it is important that the current not decay too quickly, since the dwell time behavior is a direct function of the current and resistance in the arc. A decay constant of at least 100 milliseconds is desirable. If the current is allowed to decay too fast (because of a capacitor bank with insufficient capacity), a less than realistic voltage drop will be produced along the arc channel of the important downstream portion of the test panel. Consequently, the arc may tend to stay attached to a forward point until extinguished, without reattaching or sweeping the full length of the test section.

In the MCAIR facility, the fine wire used to initiate the arc is vaporized by switching on the continuing current source. After initiation, the arc is sustained at about 400 amps by a 240-kilojoule capacitor bank discharged through a high voltage resistive element which is adjusted for the desired current level. The high-voltage resistor also serves to decouple the capacitor bank from the restrike pulse voltage when it is applied at the overhead rod electrode.

5.2 THE RESTRIKE SIMULATION. The restrike simulation is the most difficult component to simulate accurately, but it is essential to the evaluation of dielectric coatings or systems. Puncture of dielectric surfaces is a voltage-dependent phenomenon. The fast rise time of the current pulse produces an inductive voltage drop along the channel. The instantaneous voltage is $V = L\,di/dt$; the inductance, L, is determined by the configuration of the arc channel and the di/dt is determined by the generator impedance and the test circuit configuration. It is very difficult to produce a full 100-kiloamp restrike with the prescribed rise time of one to two microseconds into the high impedance test circuit required for swept stroke testing. Consequently, lower peak currents, representative of average restrike values (from 15 to 50 kamps) have to be accepted. The impact of this compromise is that tests of dielectric protective systems require extrapolation to determine threat level results.

In the MCAIR facility, the swept stroke high-peak current restrike is provided by a 0.5 megavolt Marx Surge generator which produces a fast rise time high-current pulse, reaching 50 kamps in one to two microseconds. An intermediate current component can also be coupled onto the restrike, which can be triggered at a preselected time to occur at the desired position on the test panel.

5.3 DIAGNOSTIC EQUIPMENT. In addition to current oscillograms, still photography and high-speed movies are used to monitor the arc channel attachment behavior. The electrical and photographic data, coupled with an examination of the test specimen, yield arc dwell times, dielectric hold-off efficiency and continuing current damage results. The test panels can be mounted on light-tight enclosures for filming backside sparking.

5.4 INTERPRETATION OF RESULTS. Wind tunnel facilities have been used to evaluate the baseline lightning vulnerability of new aircraft structural designs and to evaluate candidate lightning protective measures. Figures 8 through 11 illustrate swept stroke behavior on composite surfaces with various types of protection. The lines shown in the photos which join the damaged areas do not indicate the exact motion of the arc across the surface, but simply connect the arc attachment points which occurred during a single swept stroke run. To prevent local air turbulence and possible preferential reattachment of the arc to a previously damaged area, the damaged areas were covered with a dielectric material prior to any subsequent test. For the numerous tests shown, the specimen was repositioned so that the windblown arc was always initiated at the centerline of the wind tunnel nozzle exit.

Several interesting results have been produced during these tests. For example, it was supposed that there should be no reason for the arc channel to divert out of the streamline to attach to protruding objects. No diversion from the streamline was observed, but it was observed that the arc channel would preferentially pick up regions of discontinuity (such as metal bolt heads in a composite panel, or seams between dissimilar materials).

Restrikes generally cause the arc to reattach at a new point, but the reattachment is not necessarily at the knee of the arc channel where it turns up out of the boundary layer. Since it is not practical to produce a threat-level high-current restrike (100 kA) with threat-level rise

time (one to two μsec), it is necessary to extrapolate the test results on the basis of the ratio of threat-level current (defined in relevant specifications) to test level current (assuming a one to two μsec rise time is used). For example, if a restrike test of a dielectric protection system showed that a 25-kA restrike would be held off over a 24-inch span by the dielectric strength of the protective material, then extrapolation to threat level would yield a six-inch safe span at 100 kA.

It should be emphasized again that when a swept stroke fast rise time restrike is used, it can only determine attach point behavior; it gives no information on damage and incomplete information on internal sparking. However, damage and sparking tests can be conducted subsequently with a static high-current damage waveform which need not have the large circuit impedance of the swept stroke test. A 100 kA damage current test can be readily conducted at worst-case positions or areas on the test surfaces (see Figure 9).

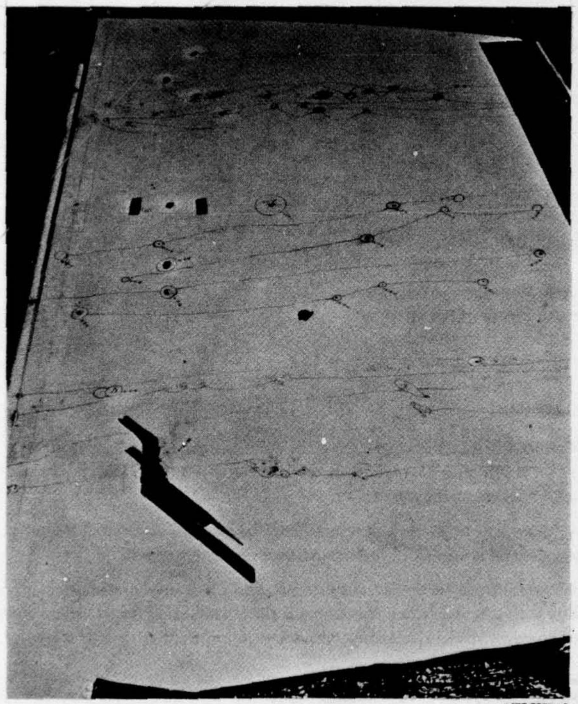

Figure 8 - View of Stub Wing After a Series of Swept Stroke and Static Restrike Tests

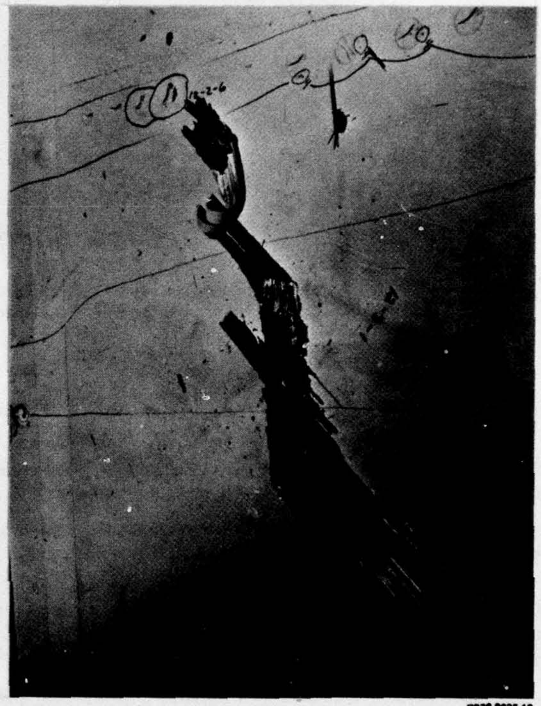

Figure 9 - Closeup of Unprotected B/E After Swept Stroke and Static Restrike Tests

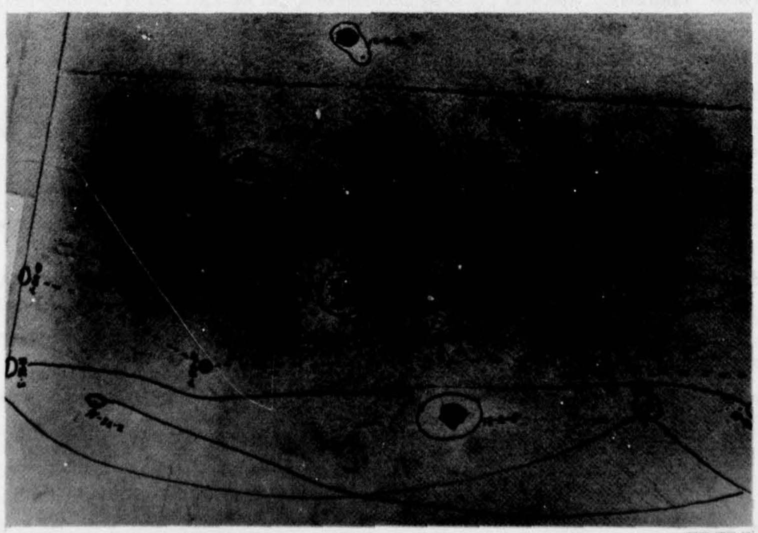

Figure 10 - Closeup of Foil Protected Area Containing Exposed Head Fastener Row After Testing

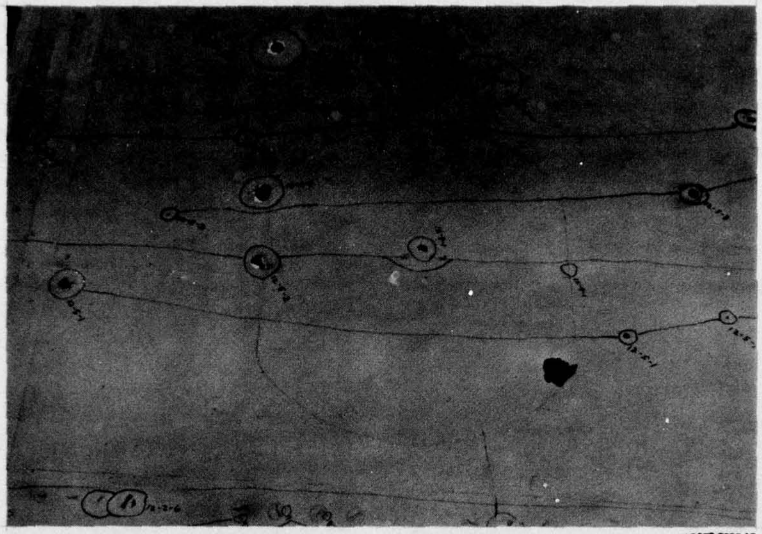

Figure 11 - Closeup of Foil Protected Area Containing Mylar Covered Door and Covered Head Fastener Row After Testing

6. CONCLUSIONS

It has been shown that a properly designed and carefully controlled windblown arc test can be used to simulate aircraft swept stroke lightning behavior. Although other techniques such as sled tests and magnetically driven arcs have been used for dwell time tests, it appears that additional development will be required before full confidence can be placed in the results.

Arc dwell time tests produce realistic burn-through damage on aircraft surfaces bearing close resemblance to in-flight damage observed on aircraft struck in service. The tests are useful in evaluating skin thickness and surface finishes in fuel-bearing structures.

Swept stroke restrike tests are useful for evaluating nonmetallic structures, dielectric protection systems, and other nonconductive surface finishes. However, it is not practical to generate full threat-level restrikes with fast rise times. Results of restrike tests should be extrapolated to determine the actual dielectric strengths of protective materials used to cover vulnerable areas or to determine the actual size of a safe area with a given nonconductive surface.

Restrike damage and sparking tests should be conducted in a static test with a full-energy damage waveform. Low quality air may be used to produce aerodynamic cooling of the surface if the longer duration intermediate current component is used.

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QUESTIONS and ANSWERS

1 – From B. Burrows

Q – Has the arc voltage drop in wind tunnel tests been measured ?

A – Not directly. However, we routinely measure the current flow through the test circuit. We are confident that the impedance of the fixed circuit elements such as waveshaping resistors, transmission lines, test article, etc. does not vary significantly with current. Therefore, we can relate variations in the durrent record directly to variations in the impedance of the arc channel, which is really the parameter of interest. For example, we can often see in the current record where reattachment of the arc occurs since the arc length is shortened abruptly, the resistance of the arc is decreased and the current increases.

2 – From A.W. Hanson

Q – Can you say what length of arc your generators could support in your swept stroke tests ? In the Culham rocket sled tests, arcs approaching a total length of 20 feet (6 meters) were observed.

A – It is not possible for me to state exactly what length of arc our generator could support because we've never attempted to measure it. Our photographic records are generally limited to the immediate field of view of the test surface. The arc obviously blows out beyond the test article several feet, but by that time the voltage has decayed on the capacitor bank to a low value. I'm sure that an analysis of the impedance of an established arc channel would show a 12,000-volt source could sustain a very long arc, perhaps even 20 feet. In our low velocity wind tunnel, we have observed arc extensions in excess of seven feet (2.5 meters) after the voltage on the capacitor bank has decayed to less than 25 percent of the initial value. A photograph of the seven-foot arc is shown below in Figure A1.

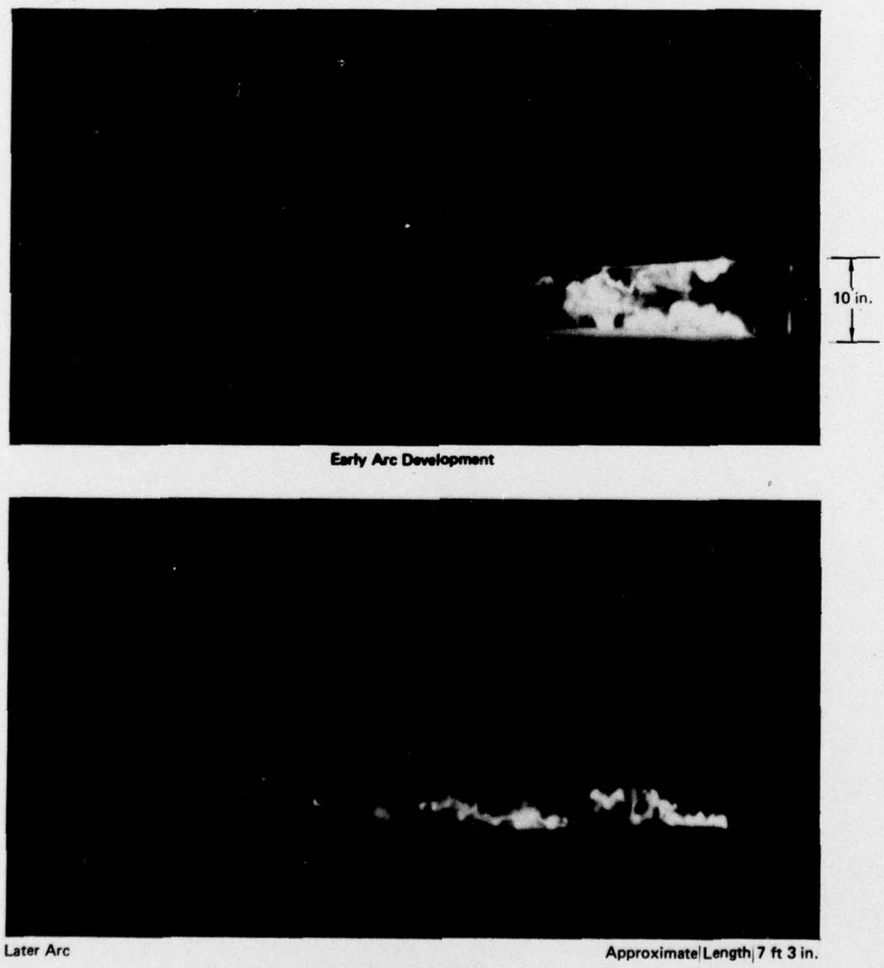

Figure A 1. Swept Stroke Arc in Low Speed Wind Tunnel

LABORATORY SIMULATION OF SWEPT LIGHTNING STROKES

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SUMMARY

Other methods of simulation of swept strokes than by the use of wind tunnels are described. The results of magnetically swept arc studies and rocket sled simulations are compared. Longer dwell times are observed with the rocket sled, and skip distance is not well correlated with dwell time. Arc instabilities appear to play a significant role in bridging the air gap between the arc and the specimen surface. A lower limit to the skip distance is set by the breakdown potential of the paint layer, if any. It is suggested that aircraft geometry and speed are not important in determining dwell times. The time between current surges in the lightning stroke should be considered as the upper limit for the dwell time.

1. INTRODUCTION

Wind-tunnel simulations of swept lightning strokes have been studied by several authors (Brick 1968, Robb et al 1970, Brick et al 1970, Oh and Schneider 1975). The air moving over the stationary surface can be a good representation of the aerodynamic conditions around an aircraft if the air flow is properly-controlled. The arc, struck to a stationary surface from a stationary rail electrode, must produce two simultaneous swept-strokes on the rail electrode and the surface. This is not realistic, though a highly-polished rod electrode allows an arc contact to slide almost continuously along. Any attachments on the rod will cause the arc current and voltage to be subject to unrepresentative changes.

Magnetically-swept arcs have been used (Hanson 1977) as an alternative method of simulation. The aerodynamic flow is not represented at all, and the arc is moving with respect to the air. The most realistic simulation can be arranged by moving a test surface through a stationary arc in stationary air, for then the aerodynamic flow and the arc behaviour is representative of a lightning strike to an aircraft. This paper discusses magnetic sweeping briefly and reports more fully on work with a moving test vehicle, a rocket-sled swept-stroke experiment.

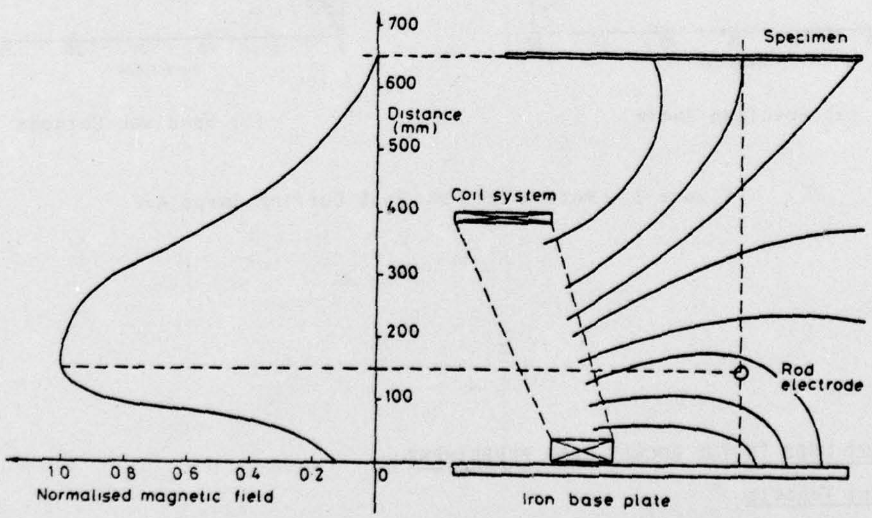

Figure 1 - Horizontal Component of Magnetic Field of Swept Stroke Rig

2. MAGNETICALLY-SWEPT STROKES

The coil system and iron base plate shown in Figure 1 was used at Culham. It produces a magnetic field which is constant at a given height over a distance of 2 m. but which varies with height within the apparatus. The coil current is driven by a 4 sec 0.5 MW pulse from a dc motor-generator set. When an arc is struck between the rod electrode and the flat plate specimen above the lower section of the arc is driven at higher speeds.

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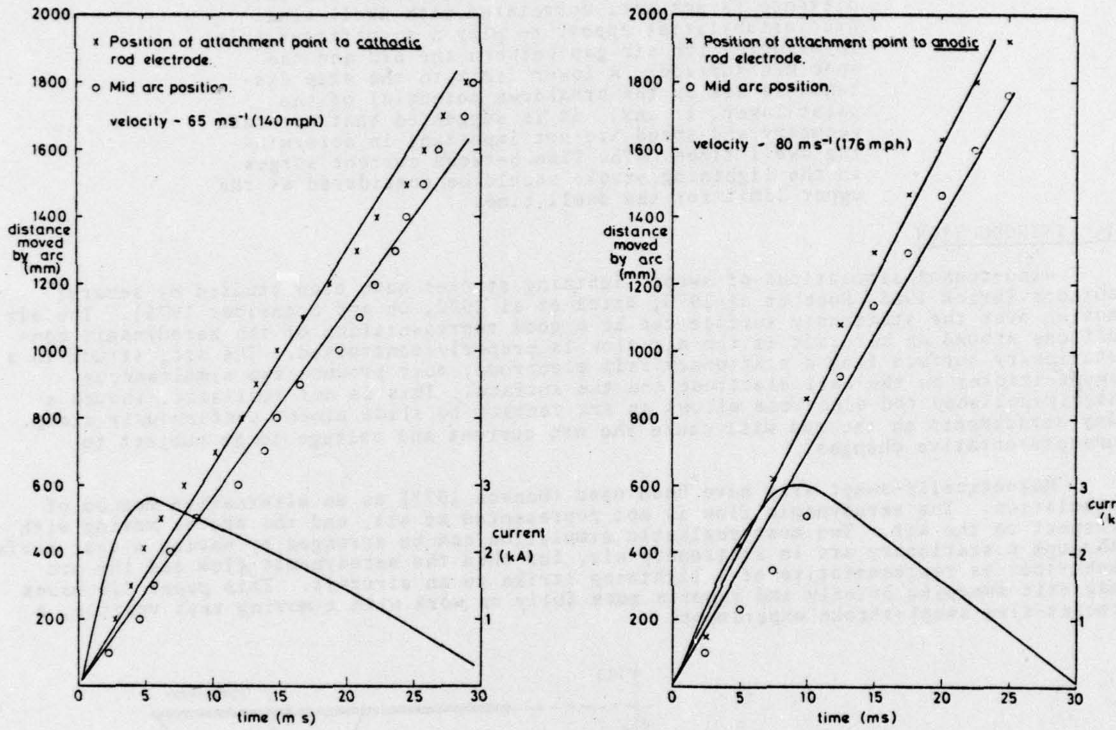
Thus a simulation of the boundary layer at the surface of the specimen is obtained.

If current is driven by a suitable capacitor bank through a thin wire bridging the gap between specimen and rod, the current pulse in the air arc initiated as the wave melts can be made unidirectional by clamping (Hanson 1977). The current I is not constant, so that the velocity due to the product I x B is a magnetic field B might be expected to vary. In practice it does not, as Figure 2 shows. The drag forces on the arc must be proportional to current in the same way as the accelerating force I x B.

The velocities achieved with about 3 kG peak field are in the region of interest to aircraft for take-off and landing: 3 kG was the value of the horizontal component of B in Figure 2. With additional windings the maximum obtainable velocity of the arc could be increased to about 300 m/s. The results are compared with other work in Section 6.



(a) Specimen Anode

(b) Specimen Cathode

Figure 2 - Motion of 3 kA Peak Current Swept Arc

3. EQUIPMENT USED IN THE ROCKET SLED EXPERIMENT

3.1 The Test Vehicle

The test vehicle consisted of two 'wings' and a support frame mounted on an existing rocket sled. The 'wings' were made from 8 ft. long demountable test sheets fixed to an aluminium frame. Advice was given by RAE on the design of the test vehicle. It was suggested that flat test sheets should be used for simplicity and because this was representative of most aircraft surfaces. Only one wing passed through the vertical arc, producing attachment points on both the upper and lower surfaces. The other wing was included in the design solely to balance the wing under test and provide aerodynamic symmetry. Figure 3 shows the vehicle (Dobbing and Hanson 1978).

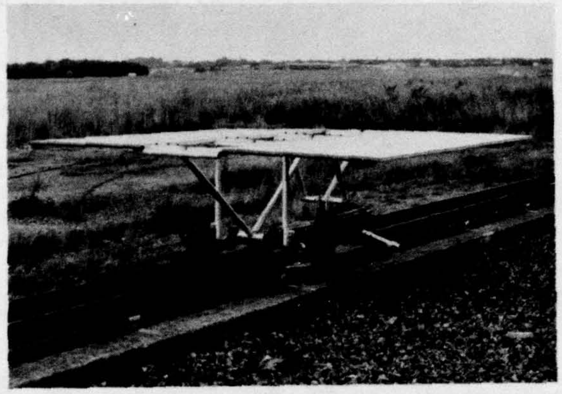


Figure 3 - Test vehicle used in swept stroke experiment.

Tests were also conducted with an air spoiler made from a 22 mm wooden quadrant attached to the leading edge. This produced a thicker boundary layer, thus simulating a longer length of wing.

Painted and unpainted metal or carbon fibre composite panels were used.

The carbon fibre panel which was fixed to the top surface of the wing for some tests consisted of three plies ($0^o/90^o/0^o$). Two plies ran perpendicular to the airstream and one ply parallel with it. The panel was 0.46 mm thick and made from Hyfil fibres. One surface of the panel had 5 μm aluminium foil glued to it. This side was then sprayed with layer of epoxy paint. The other side of the panel was untreated.

3.2 The High Current Arc Generator

The transportable inductive storage arc generating system described by Hanson (1977) was used. It produced a unidirectional pulse with amplitude and duration in the continuing current range from a lead-acid battery with an inductive storage system. Arcs more than 5 m long carrying 600 A could be produced.

3.3 Electrode Assembly

The electrodes were supported from a scaffold gantry which straddled the track. To reduce unrepresentative forces on the arc caused by the return conductors, these were distributed to give a low field region around the electrode. The electrodes were of the jet diverting type described by Hanson (1977). The electrode spacing was 200 mm.

4. EXPERIMENTS

The effect of varying the sled speed, electrode polarity changes and the effect of the spoiler were investigated using the aluminium panels with a single epoxy layer.

One of these sets of conditions was then repeated using panels having a double layer of epoxy paint, to check the effect of paint thickness. The air spoiler was not used for these tests. A panel with an extra thin layer of paint was used in the upper surface only, with the air spoiler in place. In these test bare aluminium without an air spoiler was used on the underside of the wing.

Similarly the carbon fibre panels, with and without surface protection were mounted on the upper side with an air spoiler, whilst bare aluminium, without an air spoiler, was used on the underside.

In all tests the peak current was about 600 A peak current decaying by no more than 50% during the period of attachment to the wing.

Altogether 30 runs were conducted, with two or three runs at each set of test conditions.

The arc current and voltage signals were displayed on an oscilloscope together with the time taken for the sled to pass between two position sensors. The arc was also viewed with a cine camera running at 12,000 frames per second. Subsequent analysis of the cine film was used to give the arc dwell time with an accuracy of $\pm$ 0.2 ms. Arc skip distances were also obtained from the cine film.

5. RESULTS

5.1 General Observations

The arc was initially driven ahead of the leading edge of the wing, which was made from an insulating material. This part of the test is of course not a true representation of natural lightning and no results are quoted until attachments to the wing are established. In general breakdown occured simultaneously to the top and bottom of the wing. To aid initial breakdown on painted surfaces, scratches running along the leading edge of the test sheets were made.

Once the arc had attached to the surface a jet from the attachment point would penetrate the airflow boundary layer, so that the bulk of the arc channel was in stationary air. When the air spoiler was used the arc was driven significantly further from the test sheets resulting in longer dwell times.

The arc was highly unstable, showing predominantly kink instabilities. The current, voltage and velocity diagnostics are shown for a typical run in Figure 4. The test sheets were treated with a single layer of epoxy paint and the sled speed was 52 m/s.

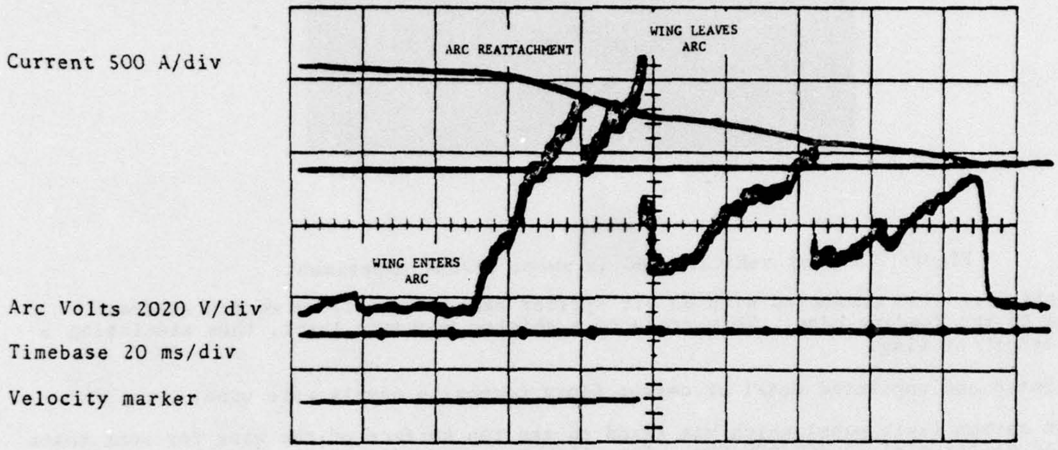


Figure 4 - Current, Voltage and Velocity Measurements

As the wing entered the arc it extended and the arc voltage rose. Thus the current decayed more quickly. When a reattachment of the channel occurred and the channel length was reduced a sharp drop in arc voltage appeared. As the wing left the arc the channel reformed behind the wing, causing another sudden drop in arc voltage. The new arc channel was drawn out in the wake of the wing until another breakdown shortened the channel again. The process terminated when the arc current fell to zero.

5.2 Results with Painted Surfaces

On the underside of the wing the arc remained close to the surface as it was extended. The voltage across the length of the arc between attachments was measured, and found to be close to the breakdown potential of the paint, which was independently measured. This supports the accepted model of the swept stroke mechanism when no air gap exists between channel and wing surface. Polarity changes had no effect, as expected.

The potential gradient in the arc was found to be 1.34 ± 0.44 kV/m.

When the air spoiler was used air-gaps appear under the wing between the arc channel and the surface. Over the wing larger air gaps appeared with and without the air spoiler; this is believed to be a buoyancy effect. Few reattachments formed on the upper surface. Only in 3 runs was an attachment formed after the first attachment to the scratch on the leading edge: in 14 runs the skip distance exceeded the wing length (2.44 m). In 2 runs no attachment at all was formed to the top surface.

The peak voltage between the arc channel and the trailing edge of the wing was 3 kV, showing that airgaps have a significant role in the swept stroke process. The breakdown voltage of the paint used (single layer) was 700 ± 200 V.

When scratches were made in the paint, perpendicular to the air flow, on the top surface the arc always attached to a scratch, but not to every scratch. The separation between scratches was 300 mm.

5.3 Results with Bare Metal

Bare metal sheets were used only on the underside of the wing. The results are shown in Table 1.

Table 1 - Tests on Bare Aluminium at 52 m/s

Polarity of Wing	Skip Distance (mm)	Dwell Times (ms)
Anode	130 ± 100	2.8 ± 2.2
Cathode	260 ± 230	6.5 ± 3.6

Skip distances and dwell times are not correlated: both are random processes. There is no variation with distance from the leading edge, but cathodes show significantly longer dwell times than anodes. Cathodes tend to show a continuous track (up to 100 mm) rather than a single arc root at each attachment.

These results suggest that arc instabilities decide the time interval between positions of close approach of the arc to the wing, and thus the random arc motion determines the dwell time.

5.4 Results with Carbon Fibre Composite (CFC) Sheets

Attachment points to plain CFC surfaces showed tufting, mostly in the two outer plies of the CFC. Current transfer was more gradual than for metal sheets, requiring 1 to 5 ms.

Sheets protected by aluminium foil gave dwell times rather less than painted aluminium sheets. The foil protected the CFC material by evaporating rapidly and causing the arc root to move over a wide area.

Figure 5 shows the arc dwell times on untreated CFC to be 5.7 ± 5 ms.

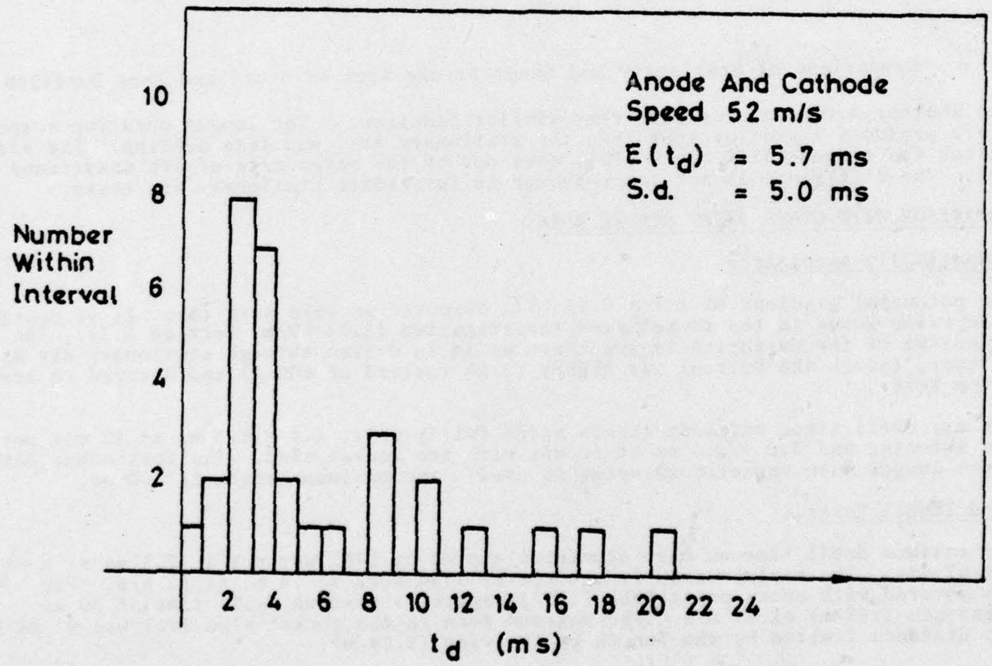


Figure 5 - Relative Frequency Diagram of Dwell Times on Plain CFC Sheet.

5.5 Comparison with Stationary Arcs on Metal

Areas of metal containing arc attachments were cut from the test sheets. These were sectioned through the centre of the arc attachment, polished and etched. Two stationary arc tests at closely the same current and duration were conducted on samples of the same sheet, using the techniques described by Hanson (1977). Two sets of results are shown in Figure 6.

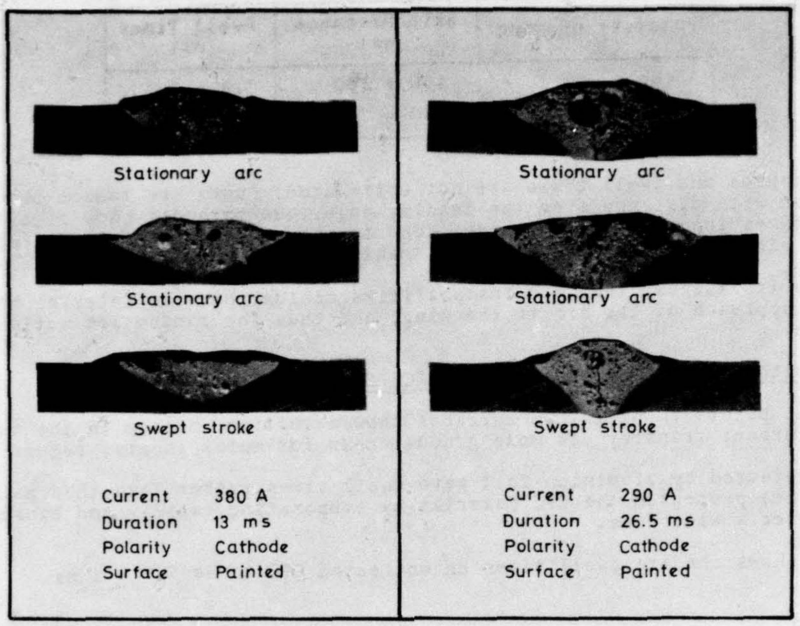


Figure 6 - Comparison of Stationary and Swept Stroke Arcs of Short and Long Duration

The shorter duration arcs show very similar behaviour. The longer duration swept stroke arc produces a smaller spot than the stationary arc, and less melting. The example shown gives the largest difference (40%) seen out of the seven sets of arc conditions examined. The difference is not large enough to invalidate stationary arc tests.

6. COMPARISON WITH OTHER SWEPT STROKE WORK

6.1 Magnetically-swept Arcs

The potential gradient of 2.4 ± 0.48 kV/m observed in this work (Sec. 2) is nearly twice the value found in the rocket sled investigation (1.34 kV/m, Section 5.2). The forced cooling of the magnetically-swept arc as it is driven through stationary air may be important, though the current was higher (3 kA instead of 600 A) and decayed to zero during the test.

The arc dwell times on anode sheets agree fairly well, 2.4 ± 1.3 ms at 62 m/s for magnetic sweeping and 3.1 ± 2.4 ms at 52 m/s with the rocket sled. The continuous cathode tracks are longer when magnetic sweeping is used; the maximum length is 500 mm.

6.2 Wind Tunnel Tests

The maximum dwell time on bare aluminium quoted by LTRl workers is 2.5 ms at a wind speed of 67 m/s. The maximum seen in the rocket sled work is 14 ms at 52 m/s. For surfaces covered with epoxy paint Robb (1977) reports a maximum dwell time of 20 ms (skip distance 1.27 m) at 67 m/s. The maximum seen in the rocket sled work was 47 ms with the skip distance limited by the length of the wing (2.44 m).

7. CONCLUSIONS

The minimum skip distance is determined by the breakdown potential of the paint (if any) on the surface and the length of the air gap. This gap length is governed by the air flow pattern, arc buoyancy and arc instabilities. Skip distances and dwell times are uncorrelated random variables.

The small electrode spacing in the rocket-sled holds the arc close to the wing, so that the measured dwell times are optimistically short. Even so on painted surfaces the work cannot set a firm upper limit for the dwell time. The limit must be above 50 ms approximately for the dwell time, and the skip distance can be greater than 2.5 m. These are greater than values seen in wind-tunnel tests.

On aluminium surfaces the rocket-sled work is in fair agreement with results from magnetically-swept arcs and wind-tunnel studies. Stationary arc testing gives results sufficiently close to swept stroke attachments to validate the stationary arc work.

The arc dwell time must be taken as the time between current surges, rather than a function of aircraft geometry and speed, when the surfaces are painted. The existing lightning data (Pierce 1972) indicate that 50% of continuing currents in ground strokes

persist for more than 130 ms, which is well above the time at present considered as an upper limit (50 ms). If the surface is bare metal the dwell time is unlikely to be more than 20 ms, and on bare CFC sheet the same upper limit would be acceptable.

Probability arguments should be used to estimate the likelihood of metal puncture.

8. ACKNOWLEDGEMENTS

The work was carried out by members of CLSU with assistance from RAE Farnborough.

The author acknowledges the support of the Procurement Executive of the UK Ministry of Defence in this work.

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QUESTIONS and ANSWERS

1 – From D. Clifford

Q – In the sled test the behaviour of the arc channel above the wing appears to be unrealistic in that the arc channel does not lie within the boundary layer. You have ascribed this behaviour to kink instabilities, but I wonder if arc buoyancy and magnetic field forces might come into play. Have you calculated what arc displacement might be expected from these forces ?

A – The arc channel is carried through the boundary layer by the electrode jet from the surface of the sheet not by the kink instabilities. There is no reason to suppose that these jets do not appear in natural lightning strikes.

Magnetic forces due to current in the sheet are very small, because the flow pattern within the sheet provides a large degree of cancellation. The field of the lower arc channel below the sheet tends to move the upper arc away from the sheet, giving pessimistic values for dwell times. Some actual attachments will suffer similar magnetic forces due to current paths in the aircraft.

Buoyancy forces and ground effects would explain the general channel behaviour actually observed qualitatively. Displacements have not been calculated.

2 – From S.O. Schneider

Q – How do you justify using the time between restrikes as the worst case (i.e. defining the longest dwell time) ?

A – In a restrike the value of di/dt is very large, so that the voltage along a length of the channel in the neighbourhood of a metal surface is expected to develop sufficient potential along its length to cause flashover to a new attachment point.

LABORATORY TEST PROCEDURES TO DETERMINE LIGHTNING ATTACHMENT POINTS ON ACTUAL AIRCRAFT PARTS (A QUALIFICATION TEST)

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SUMMARY

Lightning flashes initially attach to aircraft extremities such as the nose, wing tips or vertical fin cap. If these extremities are covered with non-metallic skins such as fiberglass radomes, wing tips, or plastic antenna fairings, the lightning flash may puncture the skin and attach to a conductor within. Conductive diverters or other means may be necessary to protect against such punctures. Whether punctures occur or not depends upon the geometry of the structure and diverter arrangement (if present), the dielectric strength of the non-metallic skin, and the rate of rise of the electric field presented by the advancing leader. A test in which this electric field is simulated and applied to a full size replica of the structure in question may be utilized to determine the need for protection or verify its adequacy. This test is also utilized to identify lightning strike zone boundaries on metallic or advanced composite surfaces, and to determine whether windshields or canopies may be punctured by re-strikes occurring in swept lightning flashes. Typical high voltage test circuits, electrode arrangements and other considerations are described.

Purposes of the Test

The test which is employed to determine the lightning attachment points on actual (full size) aircraft parts is termed the Full Size Hardware Attachment Test - Zone 1 in the new standard. Its purposes are:

1. To determine if lightning may puncture a non-metallic component such as a fiberglass radome, wing tip or fin cap; or a polycarbonate resin component such as a windshield or canopy in zone 1A or 1B.

2. To verify the adequacy of protective diverters or other measures to prevent punctures of non-metallic components in zone 1A or 1B.

3. To determine the detailed attachment points on metallic or composite surfaces in zone 1A or 1B, for the purpose of:

   A. Establishing the boundaries of zones 1A or 1B

   B. Establishing skin thickness or other protection requirements

The Lightning Attachment Process

The process of lightning attachment to an aircraft is illustrated in Figure 1.

At the beginning of lightning-flash formation, when a stepped-leader propagates outward from a cloud charge center, the ultimate destination of the flash at an opposite charge center in another cloud or on the ground has not yet been determined. The difference of potential which exists between the stepped-leader and the opposite charge center(s) establishes an electric field between them, represented by imaginary equipotential surfaces. The field intensity, expressed in kilovolts per meter, is greatest where equipotential surfaces are closest together. Because the direction of electrostatic force is normal to the equipotentials and strongest where they are closest together, the leader is most likely to progress toward the most intense field regions.

If an aircraft happens to be in the neighborhood, it will assume the electrical potential of its location. Since the aircraft is a thick conductor and all of it is at this same potential, it will divert and compress adjacent equipotentials thus increasing the electric field intensity in the vicinity itself. If the aircraft is far away from the leader, its effect on the field near the leader is negligible; however, if the aircraft is within several tens or hundreds of meters from the leader, the increased field intensity in between may be sufficient to attract the leader toward the aircraft. As this happens, the intervening field will become even more intense.

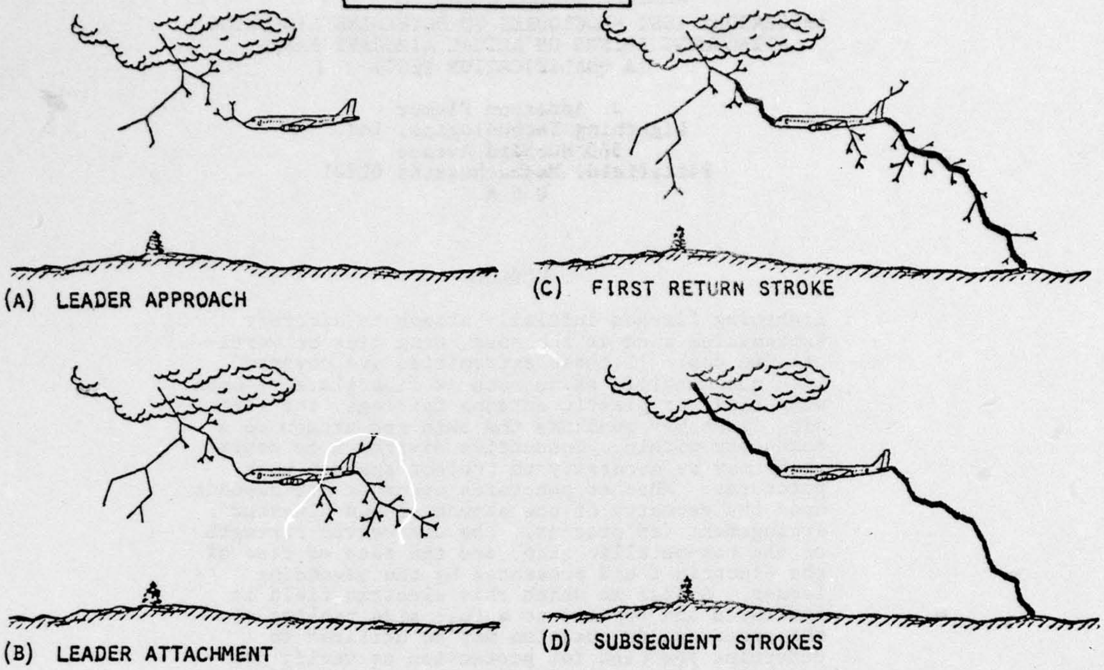


Figure 1 - The Lightning Strike Process

The highest electric fields about the aircraft will occur around extremities such as the nose and wing tips, and sometimes smaller protrusions, such as antennas or probes. When the leader advances to the point where the field adjacent to an aircraft extremity is increased to about 30 kV/cm, the air will ionize and electrical sparks will form at the aircraft extremities, extending in the direction of the oncoming leader. Several of these sparks called *streamers*, usually occur simultaneously from several extremities of the aircraft, as shown in Figure 1A. One of these streamers will meet the nearest branch of the advancing leader and form a continuous spark from the cloud charge center to the aircraft. Thus, when the aircraft is close enough to influence the direction of the leader propagation, it will very likely become attached to a branch of the leader system.

Since fiberglass and most other non-metallic materials have no electrical conductivity the electric field passes directly through them, causing streamers to originate from objects inside as well as outside of such structures. What happens may be viewed as a race between streamers propagating from conducting objects inside and outside of the nonconducting structures, as shown in Figure 2.

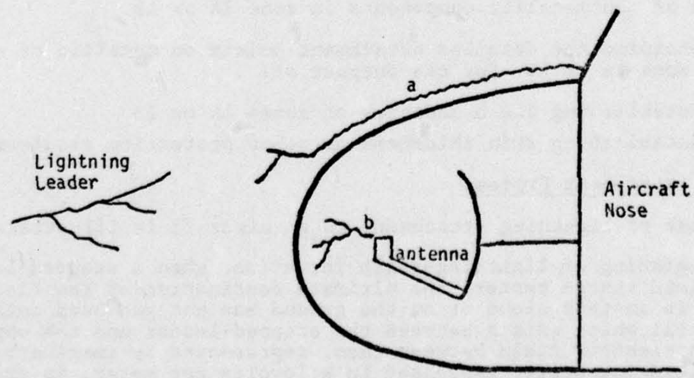


Figure 2 - Streamer Formation within Non-conducting Structures.

Both the voltage withstand capability of the non-metallic skin and the distances along alternate breakdown paths are important in establishing whether puncture or external flashover will occur. Usually, of course, a higher intensity electric field is necessary to permit the internal streamer to puncture the fiberglass wall and contact the leader than would be necessary merely to draw the external streamer through the air to the leader. In either case, as the leader approaches, the field intensity increases until one of the streamers meets the leader. The electric field about this extremity of the aircraft then collapses and the leader proceeds onward to its final destination from another extremity of the aircraft as shown in Figure 1B. When it reaches its destination the return stroke is formed as in Figure 1C, followed by continuing currents and re-strikes as in Figure 1D.

Since the purpose of the full size hardware attachment test is to evaluate initial attachment points, the electric field, and its rate of rise (dV/dt) just prior to leader attachment must be simulated. The importance of rate of rise can be explained by comparing the breakdown *timelag* characteristics of solids and air. All insulating materials, whether solids or gases, respond to high voltages and break down according to a timelag curve of the shape shown in Figure 3. Timelag effect means simply this: the shorter the time for which a voltage is applied across a given insulation, the higher the voltage must be to cause breakdown; and conversely, the longer the time, the lower the voltage necessary to cause breakdown. There is, of course, a voltage level below which breakdown will not occur at all, even if the voltage is applied for a long time. Most solids show a flatter timelage characteristic than do air or surface flashover paths, as illustrated in Figure 3.

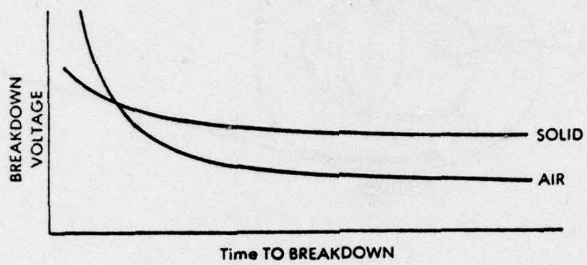


Figure 3 - Breakdown Timelag Curves for Solids and Air.

In general, an oncoming lightning flash has alternate paths to a metallic conductor, as illustrated for a radome of Figure 4. One is via a puncture of the nonmetallic skin to an internal metallic component, such as the radar dish. The other is via surface flashover to the nearest exposed metal. While the path via the puncture may be shorter than that along the outside surface, the added insulation provided by the solid skin often compensates to some degree for this, making both paths viable alternatives. The significance of voltage rate of rise now becomes evident. Because the timelag curve for the alternate paths cross each other, there are voltage waveforms that will intersect either timelag curve, as shown in Figure 4, where both a "fast" and a "slow" voltage waveform are superimposed on the breakdown timelag curves of Figure 3.

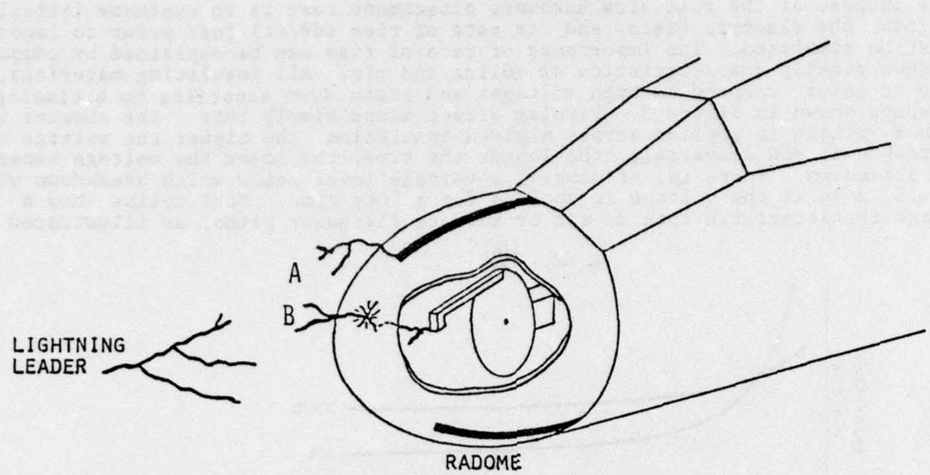
From Figure 4 it is evident that the faster rising voltage is the more severe in terms of increased probability of puncture. Thus, for attachment tests, a voltage with the fastest rate of rise expected from a natural lightning leader is desirable.

No time-domain measurements have as yet been made of the electric fields surrounding an aircraft during the lightning-strike formation process because such measurements present formidable instrumentation problems. However, it is known (Reference 1) the leader advances at about 1 to 2 x 10^5 m/s, which would result in an average of 5 to 10 μs for the leader to travel a distance of 1 m. Since about 500 kV are required to break down a 1 m air gap in 5 μs, the appropriate rate of voltage rise to use for attachment tests might be the following:

$$\frac{dV}{dt} = \frac{500\,000\text{ V}}{5\text{x}10^{-6}\text{s}} = 100\text{ kV/}\mu\text{s} \qquad (1)$$

On the other hand, if it is remembered that the actual breakdown of a single step of the leader is itself a series of smaller step breakdowns and pauses, it is likely that the rate of voltage rise across segments only a few meters long might be faster (or slower) than the average. Thus, to encompass the worst case, a rate of voltage rise 10 times as fast, or 1000 kV/μs, has been prescribed in the new standard. It is voltage Waveform A.

In practice, the timelag curves representing different paths through solid and air insulation are flatter and closer together than those drawn in Figures 3 and 4, and the point where the solid and air curves cross is less clearly defined. Thus, a relatively wide difference in applied voltage rate of rise exists between waveshapes causing breakdowns 100% of the time along one path as compared with waveforms causing all breakdowns to occur through the other path. This fact is additional support for the selection of 1000 kV/μs as the rate of rise for qualification test purposes. This waveform should create a faster (but not excessively fast) rising electric field than most natural lightning flashes create. In cases where a comparison between the results of laboratory tests using this rate of rise and subsequent in-flight lightning attachments to the same fiberglass component has been made, the laboratory tests have accurately predicted the in-flight attachment points and breakdown paths.



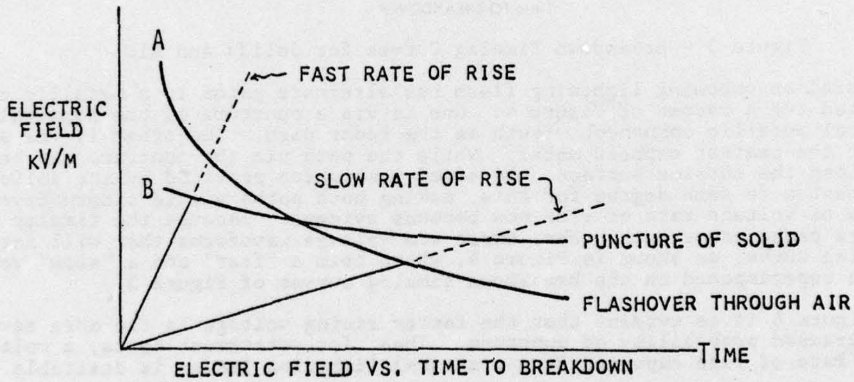


Figure 4 - How Electric Field Rate-of-Rise Determines Breakdown Path.

The Test

The full size hardware attachment test is described in the new standard as follows:

"1.0 PURPOSE

This test is performed on full size structures that include nonmetallic surfaces and is used to determine the possibility of puncture and any other paths taken by the lightning current in reaching a conductive element.

2.0 APPLICABILITY

This test method is applicable to radomes, canopies, wing and empennage tips, antenna fairings, windshields and any other assemblies located in a direct strike zone and constructed of non-metallic materials which might be vulnerable to puncture and damage from a lightning strike.

3.0 APPARATUS

The test apparatus shall include the following:

a. A high voltage generator capable of producing voltage waveform A with a peak voltage of at least 1.5 million volts
b. High voltage measuring and recording instruments
c. Photographic equipment for recording strike points.

4.0 TEST SETUP

The test object should be a production-line hardware component or a full-scale prototype. All conducting objects within or on nonmetallic hardware that are normally connected to the vehicle when installed in the aircraft should be electrically connected to ground (the return side of the lightning generator). Surrounding external metallic vehicle structure should be simulated and attached to the test object.

The test electrode to which test voltage is applied is positioned so that its tip is 1 meter away from the nearest surface of the test object. Dimensions of the test electrode are not critical. If model tests or field experience have indicated that lightning flashes can approach the object under test from several different directions, the tests shall be repeated with the test electrode oriented to create strokes to the object from these different directions.

If the test object is so small that a 1-meter gap permits strokes to miss the test object, or if a 1-meter gap is inappropriate for other reasons, shorter or longer gaps may be used.

5.0 CRITERIA TO BE SPECIFIED

a. Waveforms - Test voltage waveform A should be applied between the electrode and grounded test object.

Note: Regulatory authorities should note that additional testing using waveform B may be advisable when the test object contains flight critical components(s).

b. Number of discharges - to be fired from each electrode position.

6.0 TEST PROCEDURE

a. Set up the high voltage generator, test electrode and photographic equipment.

b. Inspect the high voltage equipment and area for safe operation.

c. Insert a dummy test object beneath the electrode, or place a conductive bar over the actual test object such that waveform checkout discharges can not damage the test object.

d. Fire a dummy discharge to the test object to check the voltage waveform and establish that the specified waveform is in fact being applied and check out the operation of the photographic equipment.

e. Place the test object beneath the test electrode and begin the tests by firing one discharge at the test object.

f. Fire the specified number of discharges from each position. Inspect the test object after each discharge and record the strike attachment points. This may be accomplished by moving either the electrode or the test object to cause the discharges to approach from each of the other direction(s) from which a natural lightning strike might be expected to approach. Repeat steps e and f. If a change results in the air gap between the electrode and the test object step d must also be repeated.

g. Tests may be commenced with either positive or negative polarity. If test electrode positions are found from which the simulated lightning flashovers do not contact the test object, or do not puncture it if it is nonmetallic, the tests from these same electrode positions should be repeated using the opposite polarity.

h. Correlate photographs with strike attachment points observed on the test object.

7.0 DATA TO BE COLLECTED

Data shall include:

a. Environmental data such as temperature and humidity;

b. Description and photographs of the test setup;

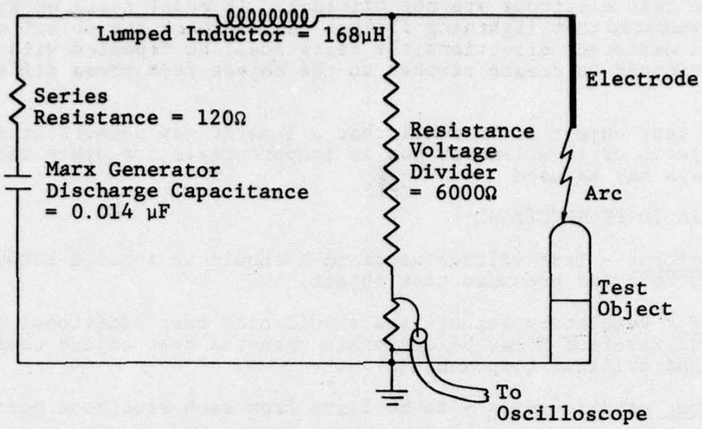
c. Date, personnel performing the tests, and location of the test;

d. Test voltage waveform oscillographs;

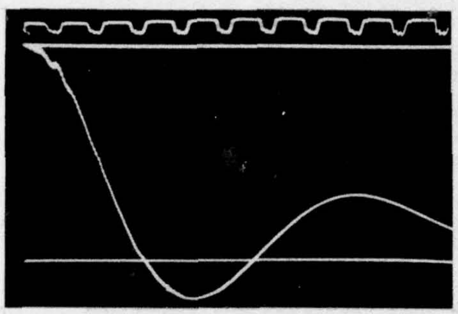
e. Photographs of discharges and attachment points on the test object.

Typical Test Circuits

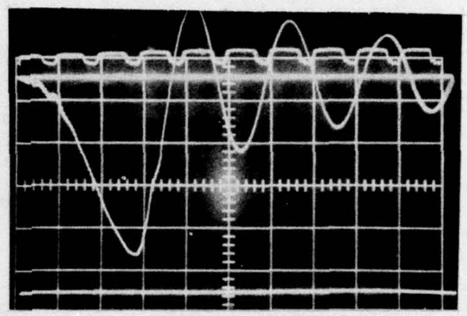
The most common application of this test is for qualification testing of a fiberglass radome or wing tip. The 1-meter electrode-to-test object spacing requires about 1300 kV to flash over. Since this must occur on the wave front, a marx-type high voltage impulse generator capable of providing at least 1600 kV is necessary. The waveshaping circuitry should provide a rate of voltage rise to 1 000 kV/μs in accordance with voltage waveform A described in paper no. 7. Typical high voltage impulse circuits that produce waveform A, together with oscillograms of the voltage they produce are shown in Figures 5 & 6.



High Voltage Test Circuit.



Crest is 1530 kV
without Test Object
in Place



Flashover to Test
Object at 1115 kV
on Rise

Voltage Waveforms
(Sweep is 0.5 μs/div.)

Figure 5 - Typical High Voltage Test Circuit
(without Load Capacitor)

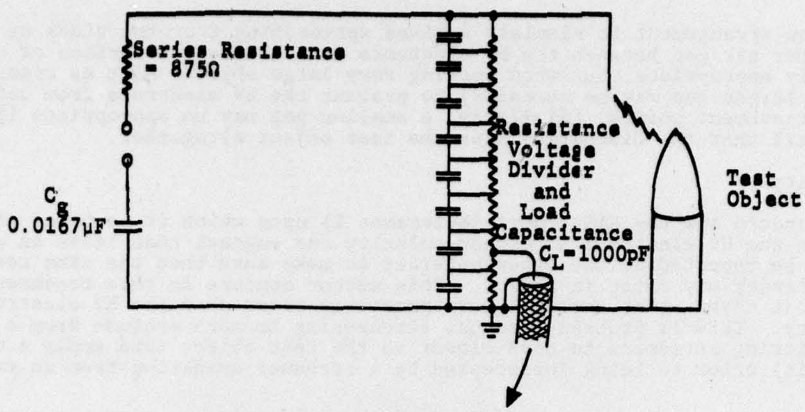
Figure 5 illustrates a circuit employing a 2 megavolt marx generator. In this circuit the stray capacitance between the high voltage circuit and earth provides an adequate load which together with the series resistance and lumped inductor controls the rate of voltage rise. The added inductance enables the voltage to overshoot the value that would be obtained with an overdamped circuit, thus permitting a smaller generator to be used than would otherwise be required. In practice, the inductor regains the 30% (or so) of rated voltage lost by normal circuit regulation. The circuit of Figure 6 utilized a 6,000 kV impulse generator. Since this far exceeded the 2500 kV needed for a test utilizing an even larger (2-meter) electrode gap, no additional inductance was necessary.

The discharge currents produced by these two circuits will not exceed 5 kiloamperes. This is sufficient to leave an identifiable mark on the test object, but not severe enough to cause extensive damage. The degree of damage that a lightning return stroke would produce can be evaluated by subsequently directing a high current discharge to the attachment point(s) determined by this high voltage test. The appropriate high current test is described in papers nos. 15 and 16.

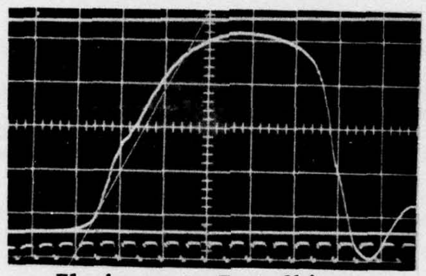
Electrode Arrangement

For qualification tests the complete test object (wing tip, radome, antenna, fin cap, etc.) is positioned in a manner representative of its location on the aircraft, with the test electrode positioned to direct strikes from each of the directions considered possible in flight. The test object should be a production-line component or an authentic replica thereof, constructed of the same materials. All conducting objects normally present on or inside it should be present on the test object. Items normally grounded to the airframe (including items such as antennas and light bulbs which are electrically close to ground) should be grounded.

If the test object is a windshield or a flush-mounted antenna normally surrounded by a conducting aircraft skin, it should be set up this way for test. If, conversely, the object is itself an appendage such as a radome, it should be set upon a pedestal representative of its location on an aircraft. Such a setup is shown in Figure 7.



High Voltage Test Circuit



Flashover to Test Object
at 2530 kV on rise

Voltage Waveform
(Sweep is 1.0 μs/cm)

Figure 6 - Typical High Voltage Test Circuit
(with Load Capacitor)

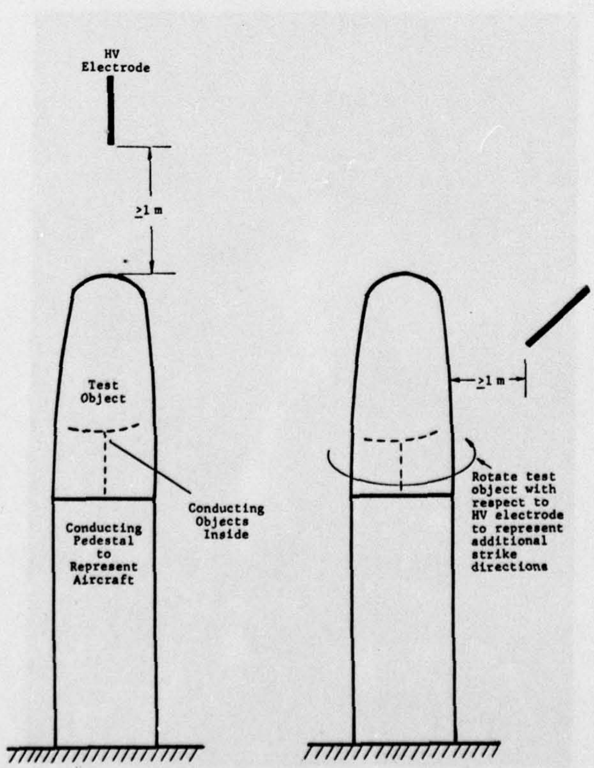


Figure 7 - Typical High Voltage Test Setup.

Figure 7 shows an arrangement to simulate strikes approaching from the sides as well as end-on. A 1-meter air gap between the HV electrode and the nearest surface of the test object is usually appropriate, but when testing very large objects such as transport aircraft radomes a larger gap may be necessary to prevent the HV electrode from unduly influencing the attachment points. Similarly, a smaller gap may be appropriate if the test object is so small that the discharges miss the test object altogether.

Electrode Polarity

The new standard and the SAE report (Reference 2) upon which it is based advise that tests begin with the HV electrode at either polarity and suggest that tests in which no punctures occur be repeated at the other polarity to make sure that the same result holds, since either polarity may exist in flight. This author concurs in this recommendation but has found, in most cases, that somewhat more punctures occur when the HV electrode is at positive polarity. This is probably because streamering is more profuse from a positive electrode, permitting streamers to move closer to the test object (and apply a more intense field to it) prior to being intercepted by a streamer emanating from an external conductor.

Number of Tests

Until recently, it was considered appropriate to apply many discharges to the test object to cover scattering effects. It is now apparent that partial breakdowns occur within some dielectrics found on aircraft, resulting in punctures after several withstands. This effect is most pronounced in fiberglass materials, including multi-ply or filament-wound laminations and foam or honeycomb-filled sandwich configurations. If such a material will withstand 3 discharges without puncture it may be considered to have passed the test, since few such structures would experience more strikes than this (at the same location) during their lifetime.

If three discharges are insufficient to evaluate scatter effects, the test object should be rotated to expose an untested surface if available, otherwise, additional test objects should be used.

Instrumentation

The inside and outside surfaces of the test object should be inspected after each discharge to identify the attachment point(s) and punctures, if any. Each point should be marked with a crayon or piece of adhesive tape so that it will not be confused with attachments produced by subsequent discharges.

In addition, each discharge should be photographed so that the path taken by the flashover can be identified. A typical photograph is shown in Figure 8.

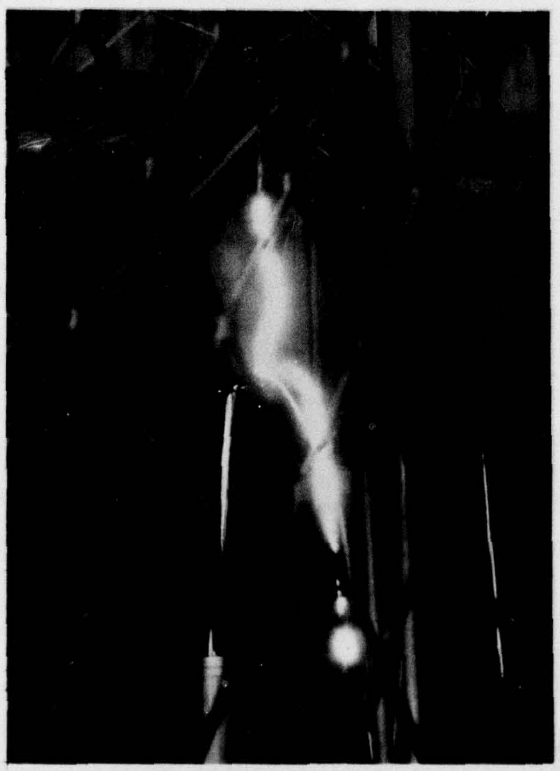


Figure 8 - Typical High Voltage Attachment Point Test of a Fiberglass External Fuel Tank.

The photograph of Figure 8 shows not only the main discharge path, but streamers emanating from other attachment points as well. Such streamers are indicative of other possible attachment points, and are visible if the test is done in a darkened room.

Re-strike Simulation

In cases where the lightning current ceases prior to a re-strike, an intense electric field similar to that which preceded the original strike may be applied to the aircraft. If the aircraft has moved forward during this time, this field may be applied to surfaces aft of the original attachment point. This possibility must be assumed to occur over any surface that may experience a swept stroke (as in zone 2A). Since the dart leader preceding the restrike may follow the original path, the electric field may be most intense about the surface beneath the channel as shown in Figure 9.

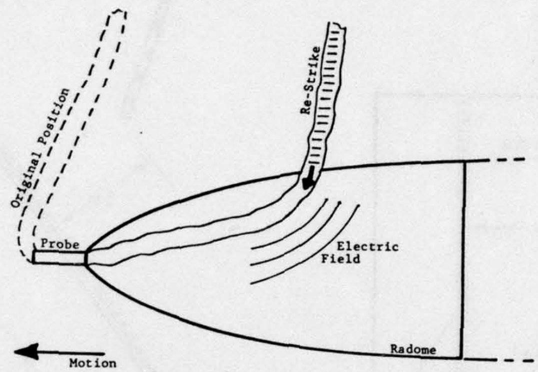


Figure 9 - Electric Field about a Re-strike.

The possibility of a puncture beneath such a re-strike can be evaluated by placing the HV electrode several centimeters above the surface, as in Figure 10. The rate of voltage rise should be 1000 kV/μs as in the initial attachment point test, but since the flashover path(s) is shorter, flashover will occur at an appropriately lower voltage.

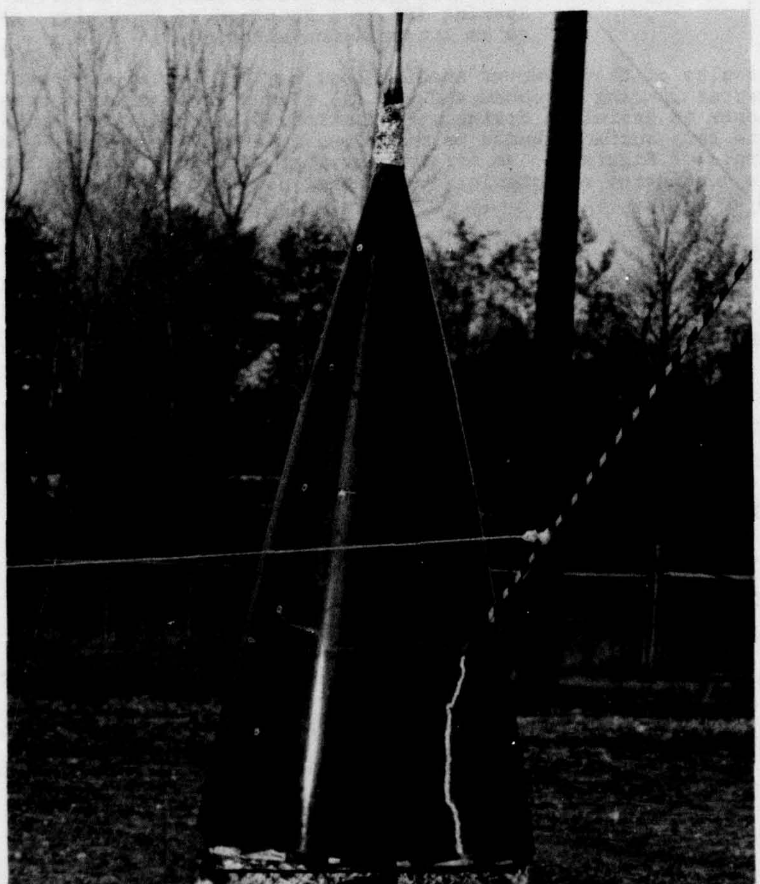


Figure 10 - Re-Strike Voltage Test.

Two flashover paths are evident in the test pictured in Figure 10. (The brighter path carried most of the current). Multiple paths are common, and care should be taken to assure that all such paths are identified. There are cases, for example, in which both a puncture and a surface flashover will occur simultaneously. A camera placed inside the test object may be utilized to identify internal streamers or punctures.

Protection Development Tests

The high voltage test may also be utilized to evaluate the performance of protective devices, such as diverter strips, or to determine the maximum separation permissible without puncture. Such tests can often be performed upon samples of the dielectric wall material using the arrangement shown in Figure 11.

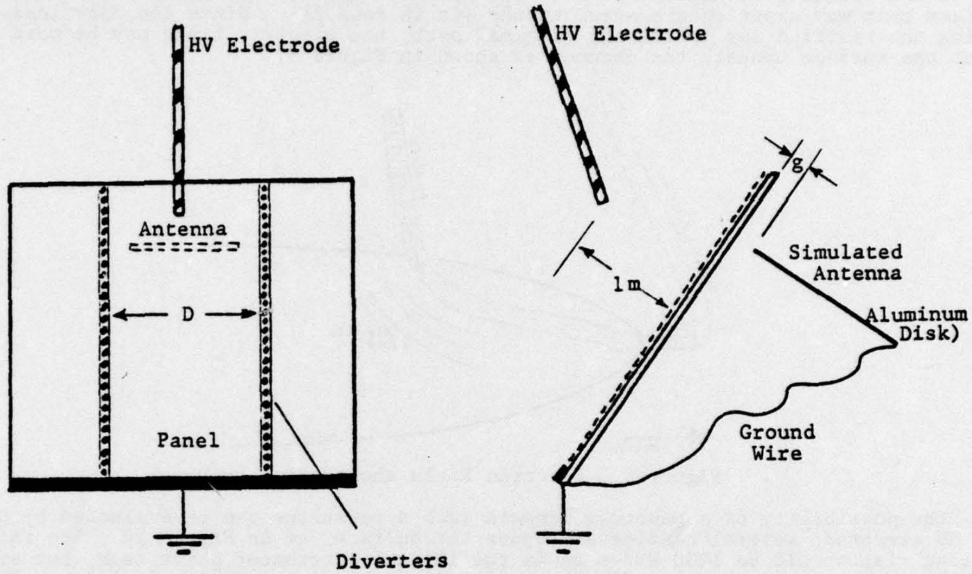


Figure 11 - Test Arrangement for Determining Diverter Spacing (D) as a Function of Proximity (g) to an Internal Conductor.

The results of the diverter spacing test may be plotted as shown in Figure 12 to relate diverter spacing to conductor gap (g) over a range of distances. Data presented in this manner is useful in design optimization. Here again, the number of discharges fired to the same surface should be minimized to avoid progressive breakdown effects. In the example of Figure 12, an arrangement that withstood at least two successive discharges was considered acceptable (i.e. having passed the test) even if puncture occurred on a third shot.

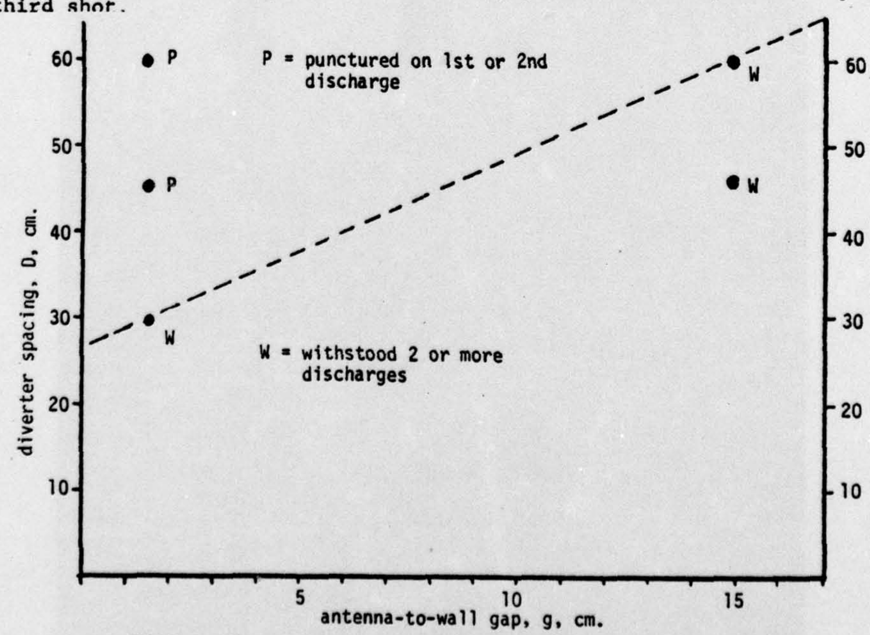


Figure 12 - Design Data from a Diverter Spacing Test.

Successful coordination of diverter spacing with internal conductor-to-wall spacing is achieved when the voltage and time necessary to flashover from any point on the external surface to the nearest diverter strip is lower than the voltage and time required to puncture the skin and attach to a conducting object beneath. The latter path includes the dielectric skin material as well as the air gap (g), but it must be remembered that the voltage withstand capability of a path through several different materials is less than the sum of the separate withstand voltages. Thus, the withstand or puncture voltage of a structure can not normally be determined by testing its components individually. The complete structure should be tested. Further discussion of the subject is presented in Reference 3.

Tests to Determine Zone Boundaries

The tests described thus far have been designed to identify puncture possibilities. The test voltage must increase rapidly (as illustrated in Figure 4) to be sure that all puncture paths are identified. The fast rising test voltage, which is provided by voltage waveform A, enables the electric field to reach sufficient intensity to puncture the dielectric skin before streamers drawn from external conductors further away reach the HV electrode. If the test voltage were increased at a slower rate, external flashovers would begin to predominate. As discussed earlier, waveform A is believed to represent the fastest rate of voltage rise produced during a natural lightning strike.

In some cases, however, it is desirable to encourage streamers from more distant, less probable locations, to identify the boundaries of a zone over which direct lightning strikes are possible. Here a slower rising test voltage is desirable. If the range of electric field rates-of-rise possible from natural lightning is as broad as the range of return stroke current amplitudes (a likely possibility since both are related to charge in the leader), voltage rates 2 orders of magnitude lower than waveform A must also be possible. Such a waveform and the high voltage test circuit that produced it are shown on Figure 13.

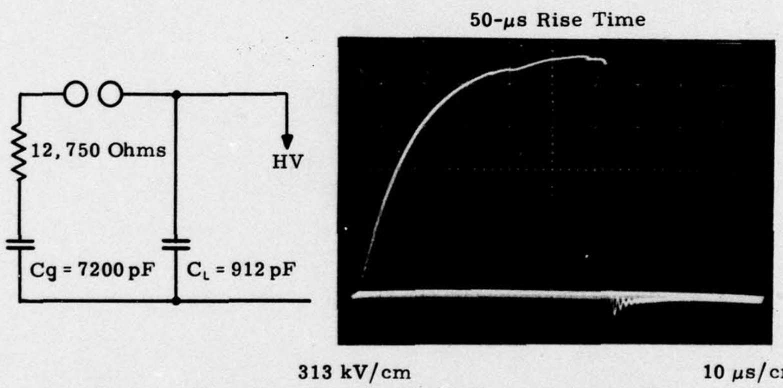


Figure 13 - Slow Rate-of-Rise Test Voltage for Identification of Zone Boundaries.

The voltage shown in Figure 13 rises to crest in 50 microseconds. Such a voltage has been identified as Waveform C in Reference 4 for use in certain engineering tests. Waveforms with rise times of up to 250 microseconds are permitted by the definition of Waveform C (50 to 250 μs) and the particular rise time is not critical. Tests with this waveform would be used to determine how far inboard zone 1 should extend on wing tips with large radii of curvature. This question is discussed further in Reference 5.

References

1. F.A. Fisher and J.A. Plumer, "Lightning Protection of Aircraft", National Aeronautics and Space Administration Reference Publication 1008, October 1977, p. 9.

2. "Lightning Test Waveforms and Techniques for Aerospace Vehicles and Hardware", report of Society of Automotive Engineers (SAE) Committee AE4 on Electromagnetic Compatibility, subcommittee SAE Committee AE4L on Lightning, June 1978, Para. 4.1.1.

3. F.A. Fisher and J.A. Plumer, pp. 207-220.

4. "Lightning Test Waveforms and Techniques for Aerospace Vehicles and Hardware", Para. 3.3.2.1.

5. F.A. Fisher and J.A. Plumer, pp. 132-134.

QUESTIONS and ANSWERS

1 – From G. Odam

Q – Have you any in-flight experience of puncture of solid radomes ?

A – Yes, assuming that "solid" means a radome wall constructed of a solid fiberglass laminate and not a sandwich construction. I have seen several radome walls of this type punctured by natural lightning, as well as by laboratory test strikes.

2 – From Ph. Lecat

Q – Is voltage waveform A intended to represent more specifically a discharge initiated from the aircraft or a stepped leader ?

A – Voltage waveform A is intended to represent the electric field appearing at the aircraft surface when attachment is imminent. As such it, supposedly, represents the field generated by the advancing leader, and this is the field that precedes (and causes) a discharge from the aircraft, and causes it to propagate outward to meet the oncoming leader.

LABORATORY TESTS TO DETERMINE LIGHTNING ATTACHMENT POINTS ON ACTUAL AIRCRAFT PARTS, (QUALIFICATION TEST)

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SUMMARY

The testing techniques for attachment point location on actual aircraft parts as proposed by the SAE committee AE4L have been comprehensively and competently described and explained by J.A. Plummer. The extent of knowledge of the underlying principles has some limitations. Some areas where additional research might be profitable are discussed and this may lead to modifications in future testing techniques. The present proposals however should give the experimenter sufficient information to enable a fair assessment of the inflight performance to be made.

1. Introduction

The present state of the art with respect to full size hardware attachment point location tests has been comprehensively and competently reported by J.A. Plummer in the preceeding paper. The author accepts the validity of the statements made in that paper, and there is no advantage to be gained in reiterating those statements merely to make relatively minor changes of emphasis of the significance of various aspects. This paper will therefore briefly look at areas where improvement in the understanding of the subject may lead to future changes and improvements in the testing techniques.

2. Streamer Formation within Non Conducting Structures

This process has been fully described by Plummer (1) in the section "Lightning Attachment Process". However, in recent work at Culham by Tallboys and Banks (2) an unexpected result was observed during an experiment incidental to the main objective of their work at the time. This result and a possible explanation suggests an area of work which may be of use in the future development of testing techniques for radome protection systems.

The DC breakdown voltage of the sphere/point air gap was first determined. The sphere was 550mm dia and the point was 6mm pointed rod. The air gap was ≈ 120mm. A thin sheet of hardboard having negligable dielectric strength was then placed in the gap at various distances from the sphere and the total gap breakdown voltage determined for each position. As expected the total system breakdown voltage did not equal the sum of the individual breakdown voltages (Plummer (1) page 12-11). It was unexpected however to find that the total system breakdown voltage could in some circumstances be well in excess of the sum of the individual breakdown voltages, and varied between the high value already mentioned and a low value of only approx 50% of the breakdown value of the original air gap. The value depended upon the position of the hardboard, and was repeatable. Fig.1 indicates the experiment and the results.

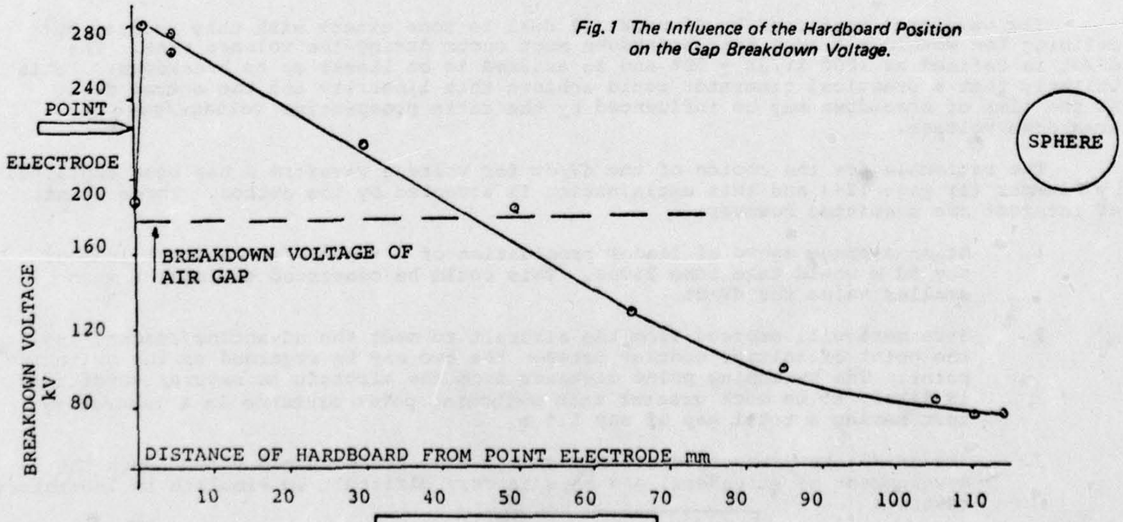

Fig. 1 The Influence of the Hardboard Position on the Gap Breakdown Voltage.

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The report gives a plausible explaination based upon the effect of charge sprayed from the point electrode collecting on the underside of the hardboard. This collected charge has the effect of changing the electric field distribution in the gap. Observations of relatively long periods of "St Elmo's fire" suggest that natural conditions do exist where considerable charge could be sprayed on the inside of radome walls prior to the final brakdown. Further work on these lines may therefore be fruitfull.

3. Choice of Electrode Size

The test methods as described by J.A. Plummer; and by many other accepted test specifications do not define the test electrode size. By implication it may be assumed that this is not important, and by general usage it may be assumed to be between say 6mm and 15mm. This is believed to be the intent of the proposed specification. Such an electrode will be placed at about 1.5 m from the test object, and the question must be asked whether this adequately represents an approaching leader say 50 m away, or even over the last 1.5 m of its final leap towards the aircraft.

Argument and counter argument can be forwarded on both sides so it is perhaps relevant to quote the results of tests conducted at ERA Leatherhead for Culham Laboratories in April 1976. These tests were designed to check the efficiency of a radome protection system. During the tests certain unexpected results were observed and in order to assess the significance of these results, electrodes of different types were used. The effects were quite marked, and in some cases the results of tests using rod electrode were the exact opposite from similar tests using a large plane electrode.

The results of course can only be fully understood when the validity of one or other method has been firmly established. There is at least two radome protection strips marketed in addition to the conventional metal diverter strips. Both appear to operate under different principles from each other and from the conventional metal divertor strip, and it may be that the testing techniques may require some modification for testing less conventional systems. Only a deeper understanding of the mechanisms involved can resolve this and clearly more basic research is required.

A further possible variant is the use of electrodes profiled to the contours of the equipotential lines existing close to the aircraft immediately before breakdown. This system was described by the author at Seattle 1976 (3). The system has been successfully employed in the investigation of dielectric failure of aqueous solutions. It has the advantage that lower voltages may be used. These lower voltages say 500/700 kV can be generated by capacitor banks giving high current up to say 100 kA. Thus both HV and High Current tests can be conducted simultaneously. It is appreciated however that more confirmatory work is needed before the system could be generally accepted.

4. Polarity and Waveform

During the previously mentioned test series at ERA (1976) differences were observed in tests using a fixed electrode system, resulting from changes in polarity, and from changes in the dV/dt. Closer analysis however showed the dependance upon polarity and dV/dt to be more apparent than real. The techniques employed to determine the voltage to be applied resulted in breakdown occuring after peak voltage for long fronted waveforms, and before peak voltage on the short fronted waveforms. Furthermore in the positive polarity tests breakdown occured at lower gap voltages than was the case for negative polarity tests. A more consistant pattern of results was observed in these particular tests, if the prospective peak applied voltage was taken as the reference rather than either the polarity or the initial dV/dt.

These factors may not necessarily have the same effect in other tests. It is nevertheless a phenomenon worthy of note and possible further investigations.

The waveforms mentioned by Plummer (1) deal to some extent with this problem by defining the waveform A such that breakdown must occur during the voltage rise. The dV/dt is defined as 1000 kV/μs ± 50% and is assumed to be linear up to breakdown. It is unlikely that a practical generator could achieve this linearity and the actual dV/dt at the time of breakdown may be influenced by the ratio prospective voltage/gap breakdown voltage.

The rationale for the choice of the dV/dt for voltage waveform A has been explained by Plummer (1) page 12-3 and this explaination is accepted by the author. Three points of interest are submitted however.

1. At an average speed of leader propagation of 2×10^5 m/s the final jump of say 50 m would take some 250μs. This could be construed to imply a much smaller value for dV/dt.

2. Streamers will proceed from the aircraft to meet the advancing leader, and the point of initial contact between the two may be regarded as the switching point. The switching point distance from the aircraft in natural conditions is likely to be much greater than switching point distance in a laboratory test having a total gap of say 1.5 m.

3. The in-flight pressure distribution may have considerable effect upon the development of streamers, and this is very difficult to simulate in laboratory tests.

Some allowance has been made for these factors in advising the use of voltage waveform B for cases where the test objects contains flight critical components, although it is felt that a considerably slower rise time could advantageously be used in some circumstances. Such circumstances arise when a test object comprised of all metal, or conducting composite components includes regions to which a strike attachment could be catastrophic (eg. a fuel vent) or could endanger the aircraft or affect its ability to carry out its mission (eg. a sensitive ariel.)

In general longer fronted waves give a wider spread of results, and some points of attachment to sensitive areas may be revealed that may otherwise have gone undetected with short fronted waves. The UK Recommended Practice (4) recongnises this by recommending the use of a 200/2000μs waveform for such circumstances.

An alternative method of identifying such potential but low probability hazards has been included in the SAE proposals in the form of the Corona or Streaming Test. Method No. 303 which is the subject of a subsequent paper at this conference to be presented by J.A. Plummer (5). It should be noted however that the comments on in-flight pressure distribution also apply to corona tests, and possibly to an even greater extent than they do to long fronted waves.

5. Conclusions

There is always some limitation in the extent to which an in-flight natural lightning strike can be simulated in the laboratory, and recognition of this must be made when specifying tests and interpreting results.

It is important that factors which could seriously affect the safety of the aircraft, or appreciably reduce its effectiveness, should be clearly identified. It is equally important however to avoid overtesting which may result from too rigid a test specification, particularly in the area of HV testing where present knowledge is still so limited. It may probably be advisable to classify such tests as "Engineering" tests rather than mandatory, thus giving the appropriate certification authority the flexibility of interpretation appropriate to the aircraft and its purpose.

Finally, in the light of our present knowledge the use of waveform A for dielectric puncture tests, and the use of waveforms A and B together with corona and streaming tests for determining attachment point locations on conducting materials will give useful information, which, if interpreted with caution, will assist the experimenter in making a reasonable assesment of the in-flight performance of the component under test. However future research work on testing techniques, particularly in the area of electrode shape and size may result in the need to modify the testing requirements.

Acknowledgement

The author wishes to acknowledge the support given by the Ministry of Defence for the work conducted at Culham Laboratory.

References

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2. J.G. Tallboys and C. Banks. Culham Lightning Studies Unit. Memo No. 67
3. A.W. Hanson. Recent Development in Testing Techniques IEEE Conference Seattle 1977.
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5. J.A. Plummer. Laboratory Tests to Simulate Lightning Streamers at Apertures (Qualification Test). Closed NATO Conference. ONERA Paris Sept 1978.

QUESTIONS and ANSWERS

From Ph. Lecat

Q – You told us of the effect of dE/dt on the efficiency of current diverters. Don't you think that dI/dt has a great influence too ?

A – Yes, but at a different phase of the capture process. The reference to dE/dt was made in respect of the probability of capture, and it can influence what is happening in the processes leading up to capture. The question of dI/dt only arises after capture and is linked with the return stroke current. Then of course it can influence where the current can flow and what the results of that current may be.

BIOGRAPHICAL NOTE

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Mr. Hanson is currently manager of the Lightning Studies Unit at Culham Laboratory. At the end of the second world was he was appointed to the British colonial service and spent nearly 10 years until 1955 in Nigeria, West Africa. On returning to UK, he joined the UK Atomic Energy Authority at Aldermaston and later at Harwell where he worked on the well known Zeta experiment. He joined Culham Laboratory in 1964 and worked on the construction of plasma physics machines until 1972 when he joined Mr. J. Phillpott in founding the Lightning Studies Unit.

Introduction to Conference Session on Lightning Attachment and Swept Stroke Testing

J.D. Robb

Two important aspects of lightning protection development are the determination of the points where lightning will contact an aircraft and the surface areas where lightning will be swept by the airflow. This information is needed for intelligent application of lightning protection measures.

Attachment point testing remains a subject of some differences of opinion in regard to basic mechanisms, specifically the electric field rise time just previous to a strike and as a result there remain some differences as to the optimum rate of rise of applied voltage in the attachment point coating. These are not academic differences as the results of attachment point testing can have real impact on aircraft design in terms of skin thicknesses and weight. The aspects of basic mechanisms and in-flight experience must always be balanced in the final analysis as will be discussed in this conference session.

Swept stroke lightning testing is of great importance in the application of lightning protection as often the presence of windstream substantially reduces the protection requirements. It is usually felt to be as important not to overapply lightning protection as it is to be sure that no hazardous configurations have remained unprotected.

It should be noted of course that testing with stationary arcs is conservative but possibly over severe. Swept stroke testing techniques are therefore of great importance in providing realistic component testing which does not unduly penalize new designs.

LABORATORY TESTS TO DETERMINE THE POSSIBILITY OF IGNITION OF FUEL VAPORS BY LIGHTNING

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SUMMARY

Of concern in new aircraft design is the possibility of fuel vapor ignition caused by lightning strikes. Lightning energies can enter the fuel tank vapor areas by sparking of joints and access doors, and by electric field coupling through plastic doors. The energies can be inductively or conductively coupled through fuel quantity probe wiring and can also cause ignition by hot spot heating of the fuel tank wall or by direct puncture of the skin. Tests have been devised to determine the adequacy of new fuel system designs and include: (a) photographic viewing of tank interiors to check for sparking or streamerings, (b) use of confined flammable mixtures about the component being tested to check directly for possible ignition, (c) infra-red or thermocouple measurements of hot spot heating, and (d) voltage and current measurements of transients on wiring.

Whereas some areas need further investigation, particularly with the new metal and composite materials being used for fuel tank skins, the fact that few aircraft losses have been caused by lightning ignition of aircraft fuel systems indicates that good confidence can be placed in the present test methods.

1.0 INTRODUCTION

One of the concerns in the design of new aircraft is lightning protection for the aircraft fuel system. Flammable vapors from aircraft fuel require a fraction of a millijoule for ignition and currents as low as 200 microamperes. Natural lightning strikes to aircraft can have crest currents of 200,000 amperes and suitable measures must be taken to assure that the large energies of the natural lightning discharge cannot be coupled into the fuel tank where ignition might take place.

These measures include careful attention to bonding practices and fuel tank joint conductivities particularly with the integral fuel tank construction. It also requires protection measures for preventing induced or conducted voltage or currents on internal fuel tank wiring also for preventing penetration of lightning strike energies through fuel vents or access doors.

The most effective way of providing economical fuel system protection is through design modification in the preliminary stages of the aircraft design process. During this phase "flip of the coin" type decisions can often be utilized to provide protection with small penalty to the vehicle in terms of weight and cost. In contrast, protection provided after the basic design has been frozen can often be relatively expensive or provide less effective lightning protection. In the following sections are presented some basic information on fuels and lightning as related to fuel vapor ignition for general background followed by a discussion of lightning testing required for design verification.

2.0 IGNITION MECHANISMS

2.1 Combustion Theory

When a flame passes through a stagnant fuel-air mixture, a process takes place which converts the cool unburned gas into the hot combustion products at a moving interface, the "combustion wave." The heat released in the flame zone is conducted and radiated into the unburned gas, raising the next layer of unburned gas to a temperature sufficient to heat this zone. This rate of heat transfer then sets the velocity of the combustion wave, which leaves behind combustion products at the flame temperature. As the fuel-air ratio is changed to increase the flame temperature, the temperature gradient steepens and flame velocity increases. In a plane combustion wave, if the flame temperature is reduced, a point is reached where insufficient heat is conducted into the next layer of unburned gas to fully ignite it, the flame stretches, and finally ceases to propagate. This is called the combustion limit, and describes the conditions under which the combustion wave is just able to propagate through the fuel-air mixture. The combustion limits of typical aircraft fuels are shown in Figure 1a.

This discussion of the plane wave may be extended to a spherical combustion wave. The ignition usually takes place at a specific point with the combustion wave moving out into the unburned gas as a spherical wave. Therefore, each new layer of unburned gas is of larger volume than the layer of combustion products which supplies the heat. When the combustion sphere is very small, the increase in volume of the next layer is so great that a mixture which would support a plane combustion wave extinguishes the tiny spherical wave front by quenching it. Therefore to obtain stable ignition, sufficient energy must be deposited at the ignition source in order to establish a flame zone of critical minimum size. (1)

In the process of fuel vapor ignition by electric sparks a very concentrated source of energy is released in the unburned gas over a very short period of time. The gas in the immediate vicinity of the discharge is raised well above the ignition temperature and an extremely steep temperature gradient is formed. As the flame zone grows, the temperature gradient becomes flatter as the excess heat deposited by the spark is added to the heat of combustion being conducted out through the surface of the combustion wave. If sufficient heat and energy have been deposited from the spark, the minimum combustion flame size will have been reached and ignition will occur.

The concept of a minimum ignition energy is of importance in lightning protection as it suggests that the mode and fine structure of the energy input time history is not important unless it is very slow and that the critical factor is in the total heat input into the spark. The details of the plasma chemistry are of course extremely complex. The gas can have electron and molecular temperatures in the form of ionization, dissociation,

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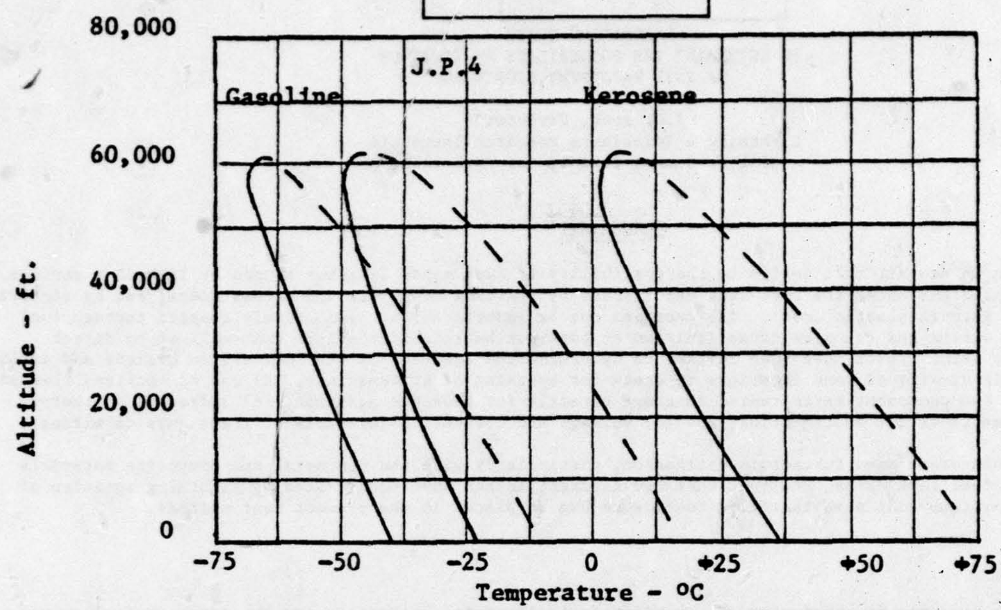

Figure 1a Equilibrium Explosive Limits Within Tanks Containing Various Fuels And Typical Fuel Temperatures, (Approximate)

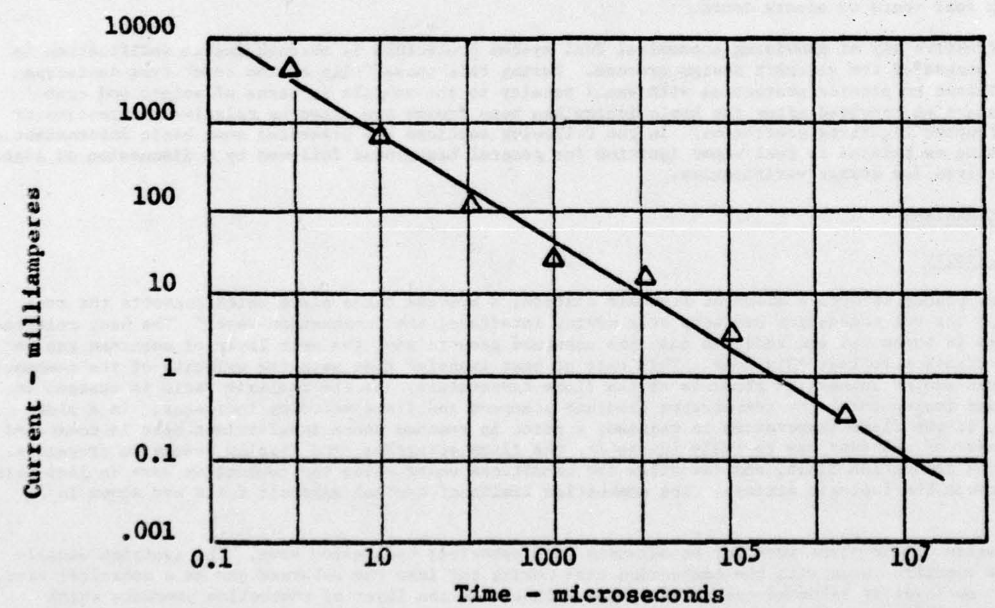

Figure 1 b Plot of minimum current vs. time duration of exponentially decaying pulse from capacitor source for ignition of stochiometric mixture of aviation fuel.

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vibration, rotation and displacement energies for the various types of plasmas such as the glow and arc types. In addition there can be ignition by conduction from contact with hot surfaces or particles.

Thus, the concept of simple heat input after some minimum equilibration or thermalization time certainly simplifies analysis of an inherently complex problem.

Another problem arises however in the fact that the heat input is not generally determined in terms of the energy. Sparking in fuel tanks is more often determined analytically or experimentally in terms of the current waveforms and a profile of the time current magnitudes for ignition is required. A rough curve of minimum ignition crest currents as a function of time (2) is shown in Figure 1b. Needed is a more careful measurement of the minimum time current waveforms for use in analysis and design of new fuel systems.

2.2 Practical Ignition Mechanisms

In spite of the excellent conductivity of the aluminum skin and the generally good conductivity of the skin joints, lightning energies can penetrate into the fuel tanks and cause fuel vapor ignition. Among the ways in which this can occur are sparking of access panels such as filler caps, access doors, drip sticks, quantity probes, pumps and drain valves (3) as illustrated in Figure 2 for a filler cap.

Lightning energies can also enter on unshielded wiring such as fuel quantity probe leads. It should be noted that fuel quantity probe wiring is often shielded but that the shield is most often carried into the tank interior without being electrically bonded to the tank wall at the entry point with the result that induced voltages on the shield or inner conductors can spark to the airframe or ground electrode on the fuel quantity probe inside the tank to cause ignition. Skin joints can also spark from lightning current flow particularly with the wet sealed joint assembly used for corrosion control and special measures are required.

The sparking inside tanks can be spark plasmas. It can also be showers of hot metallic particles. Ignition can also occur from fuel tank skin hot spots caused by lightning attachment to the outside of the fuel tank skin. With the introduction of the new non-metallic composite materials, new problems are introduced by the greatly reduced conductivity and shielding.

It is generally accepted that the minimum sparking voltage regardless of geometry and air density (Figure 3) is approximately 350 volts, and that with the excellent joint conductivities of fractions of a milliohm, straightforward analysis would indicate a very low probability of arcing inside the tank. Internal impedance drops across joints, because of skin effect, require huge external current densities to develop potentials at this level. It may be noted that typical aircraft resistances from extremities, wing to wing or nose to tail are of the order of 100 micro-ohms for aircraft ranging from the piston engine era to modern jet airliners.

However, the typical sparking that is developed within a fuel tank is not generally that which results from impedance voltage drops across the internal joints but rather that which is a result of current densities through small paths across the joints in which the current density exceeds the fusing current density. Small fingers or slivers of aluminum with insufficient cross sectional area to carry the localized current density are vaporized or melted and exploded into small spark showers. These can be generated across joints with even low joint impedances as illustrated in Figure 4. Thus, the edge treatment of wing planks and splice plates is critical in preventing sparking inside fuel tanks.

Also critical in preventing sparking inside integral fuel tanks is the type of fastener and fastener treatments. Fasteners are required which have good conductivity to the wing plank.

Fuel tank sealants help in suppressing sparking but certain minimum sealant thicknesses are required and quality control becomes critical if sealants are depended upon for sparking suppression.

Ignition can also occur from fuel tank skin hot spots caused by lightning attachment to the outside of the skin. The amount of heat transferred to the internal hot spot on the tank is determined to a large degree by the amount of time that the lightning arc remains attached to the external skin as well as the amount of energy and charge transfer in the lightning discharge. (4) It has been found that surface treatment, boundry layer and a number of other factors are important in determining the time duration of stroke attachment on the mid chord areas of fuel tank skins. (5) (6) As most of the charge transfer in a natural lightning discharge occurs in the low current long duration continuing components, the time duration of arc hangon becomes quite critical. Significant amounts of charge transfer can also be transferred in single high current restrikes, however, and for this case external skin surface treatment again becomes critical in determining the concentration of arc attachment. Unpainted skins can have a wider scatter of attachment points whereas aircraft paints tend to concentrate the arc in a small local area which greatly increases the probability of hot spot ignition or burn through.

Lightning stroke contact with integral fuel tank skins is established only in swept strike zones where the lightning sweeps over the skin, as fuel tanks are not or should not be located in direct strike zones, particularly at locations where long duration arc hang-on is possible such as the trailing edge of a section. The mechanisms of stroke contact heating are complex, involving plasma contact with the skin materials, ion bombardment and ohmic heating of the skin modified by heat transfer with phase change and windstream cooling.

With the introduction of insulating and conducting composite materials new problems are introduced by virtue of the reduced conductivity and shielding.

3.0 PROTECTION MEASURES

3.1 Skin Joints

With the introduction many years ago of high strength alloys for wing planks which formed the skins of the integral fuel tanks, a serious corrosion control problem developed because of the particular corrosion susceptibility of these materials. Wet sealed joints in which the fuel tank coating was applied wet to the fasteners and joints previous to assembly was used to control corrosion but the corrosion control engineers' desire to maintain an inherently insulated joint was completely in opposition to the lightning requirements of a good

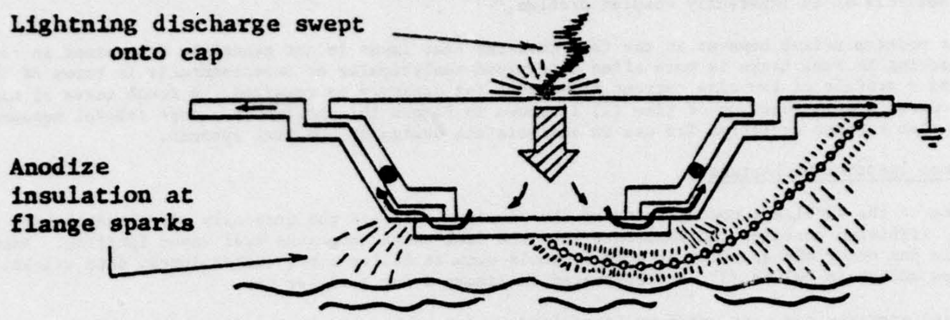


Figure 2. Fuel Cap Arcing Inside Tank at Faying Surface and on Metal Chain from Stroke Swept Onto Cap .

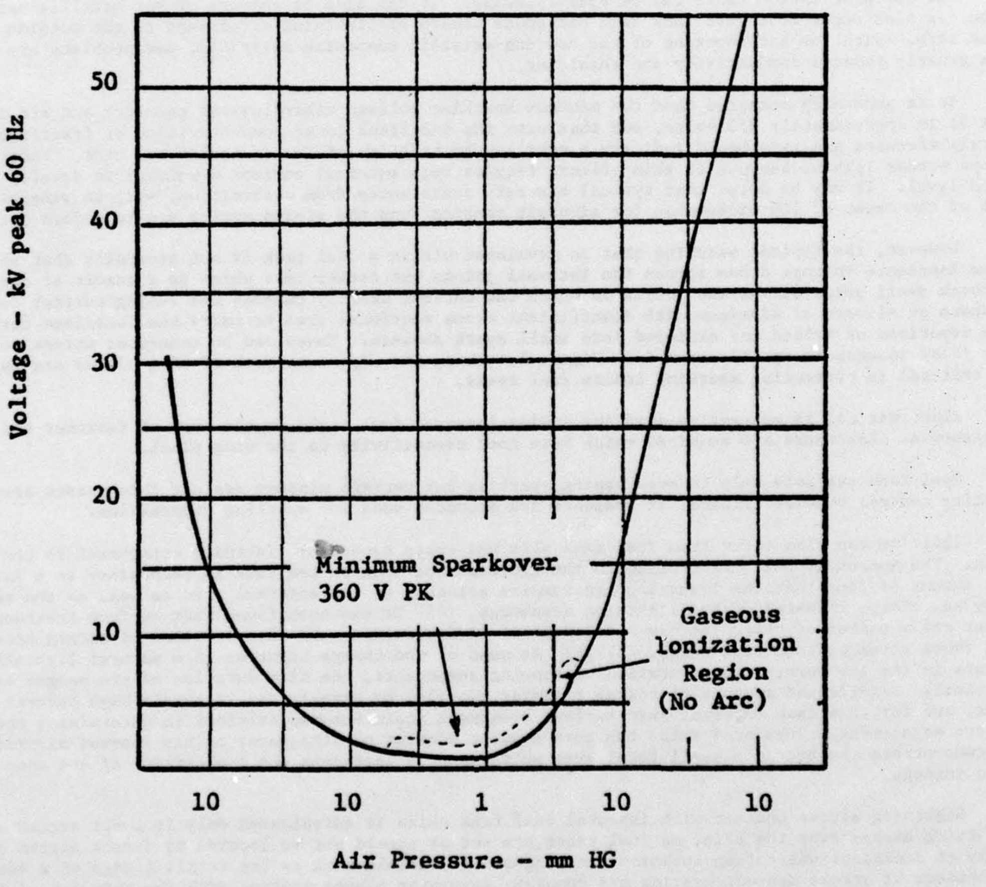


Figure 3. Rod Gap Breakdown - Voltage as a Function of Air Pressure.

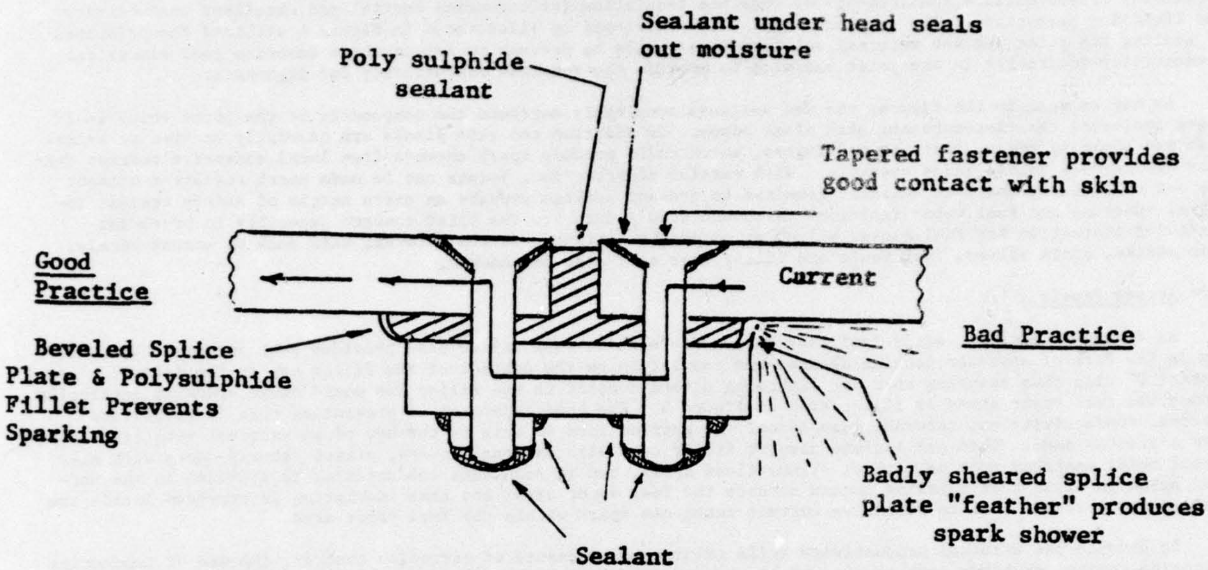


Figure 4. Wet joint lightning current paths.

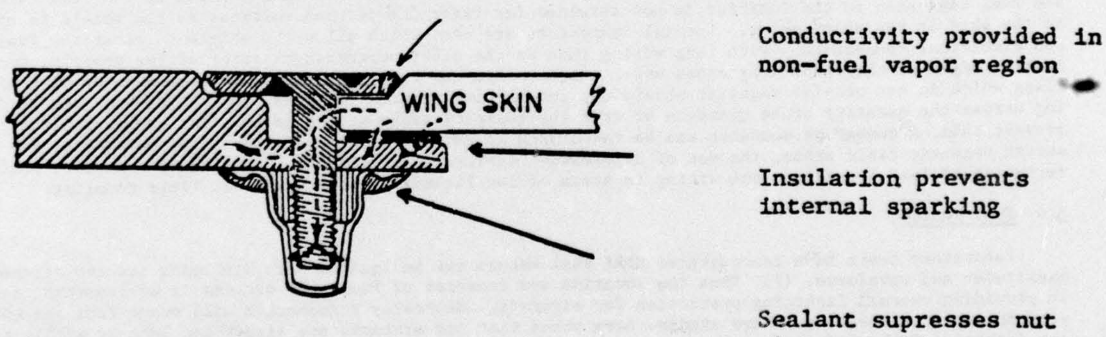


Figure 5. Lightning Resistant Access Door Provides Internal Insulation in Fuel Vapor Region and Conductivity in Non-fuel Area.

electrically bonded skin joint. Therefore, a development program was initiated to develop joints which met the apparently irreconcilable requirements of complete insulation for corrosion control and excellent conductivity for lightning protection. The resultant skin joint developed as illustrated in Figure 4 utilized the principal of sealing the joint against external moisture penetration to prevent corrosion while assuring good electrical conductivity internally in the joint sandwich to provide the required conductivity for lightning.

As may be seen in the figure, the wet sealants completely surround the components of the joint which could spark including the fasteners and skin plank edges. In addition the skin planks are carefully beveled to assure that any shearing edges, feathers or fingers, which could produce spark showers from local excessive current density are removed before joint assembly. With careful construction, joints can be made spark resistant without the use of sealants and then sealants required to prevent leakage provide an extra margin of safety against internal sparking and fuel vapor ignition. Although skin joints are the first concern generally in providing lightning protection for fuel tanks, all other components attached to the external skin such as access panels, drip sticks, drain valves, fuel vents and filler caps must also be checked.

3.2 Access Panels

As discussed earlier, early fuel tank components such as fuel filler caps provided poor internal conductivity in the form of anodized locking flanges and insulation on the outside of the filler cap in the form of a rubber "O" ring thus assuring that any lightning stroke contact to the filler cap would cause sparking internally inside the fuel vapor areas as illustrated in Figure 5. The basic approach to preventing this is to provide external conductivity and internal insulation. An extreme form of this is the use of an external metallic cover over a plastic door. This can include plastic filler caps with external covers, plastic access doors with external metal coatings such as foil or plasma flame spray, but in each case conductivity is provided in the non-fuel areas in order that sparking occurs outside the fuel vapor areas and that insulation is provided inside the tank areas to assure that no resistive current paths can spark within the fuel vapor area.

To provide the external conductivity while gaining some measure of corrosion control, the use of conducting corrosion control coatings such as the chromic conversion coatings are used while solid insulations or sometimes fuel tank sealants are used on the internal surfaces to suppress any possible sparking. This must be done for all types of fuel tank components which are installed in the wing skin.

3.3 Internal Wiring

Fortunately nearly all electrical wiring located inside aircraft fuel tanks is enclosed inside metallic tubing to assure that any electrical circuit faults and possible arcing will be isolated from the fuel vapor area. However, one important type of wiring is not shielded generally inside the tank and this is the fuel quantity probe. Although shielding is often provided on the outside of fuel quantity probe wiring, the measurement accuracy is seriously affected by the shield bonding and consequently most quantity probe wiring shields are bonded only at the indicator in the cockpit. What appears to be a shielded wire running to the fuel tank and through the fuel tank skin to the interior is not shielded for lightning induced voltages as the shield is not connected to the skin at the entry point. Special connectors are used which allow the shield to enter the fuel tank without electrical connection. With long wiring runs on the electromagnetic exterior of the vehicle, such as along leading edges of the front wing spars under non-metallic covers or along the trailing edge of wing spars behind flaps which do not provide magnetic shielding, substantial induced voltages can result which can produce sparking across the quantity probe elements or from the quantity probe wiring to the wing structure (Figure 6). To prevent this, a number of measures can be taken such as additional shielding over the outside of the wiring in strong magnetic field areas, the use of suppression devices such as small lightning arresters or preferably the rerouting of fuel quantity probe wiring in areas of low lightning electromagnetic field coupling.

3.4 Fuel Vents

Laboratory tests have demonstrated that fuel vapors can be ignited with lightning induced streamer current magnitudes and waveforms. (7) Thus the location and geometry of fuel vent outlets is an important consideration in providing overall lightning protection for aircraft. Generally streamering will occur from the outside of the vent in the free airstream where studies have shown that the mixtures are either too lean or within the quenching distance of the skin and as a result streamer ignition from effluxing fuel vapors out of vents is in general of low probability. (8) However, fuel vents could be designed which cause lightning induced streamering within the flammable vapor area and this could result in fuel vapor ignition. The general mechanism is illustrated in Figure 7. An approaching step leader induces streamering off most vehicle extremities because of the intense electric field between the step leader and the vehicle. The streamering as illustrated in a laboratory demonstration in Figure 8 is determined not only by the geometrical configuration of the aircraft and the local geometry of the vent outlet but also by the time history of the streamering from the extremities which in effect tend to provide a degree of shielding for the lower gradient areas. Thus the location and shape of fuel vents is extremely critical in determining the degree of streamering which can occur under lightning incident electric fields and the location of the streamering in the effluxing flammable vapors or in the non-flammable windstream.

The critical gradient at which vent streamering occurs can be determined using electrolytic tank plots in spite of the complex geometry which defies closed form analysis. Using a sequence of three dimensional plots for the aircraft, the local area geometry, and the vent itself, a threshold voltage can be determined for a specific vehicle height over the ground at which streamering will initiate from the vent surfaces either in the flammable mixture area or external to the vent. We are suggesting that because the tank plot approach is relatively simple, some type of vent threshold potential criteria be applied to aircraft to provide some control of vent streamering thresholds. As an example of the magnitudes, a tubular vent located at a wingtip of an aircraft may require only a few megavolts on the aircraft to cause streamering from the vent outlet whereas inboard fuel vents of the NASA reentrant Ram type may have thresholds as high as 100 megavolts. The wingtip vent threshold is sufficiently low that streamering will be occurring almost continually in thunderstorm regions from the electrical cross fields within the thunderstorm without lightning strikes even contacting the vehicle. The reentrant vents in inboard locations on the wing will have a low probability of streamering even with direct lightning contact to the aircraft at nearly any location except near the vent outlet. To apply this concept, the only problem would be in selecting a reasonable threshold potential.

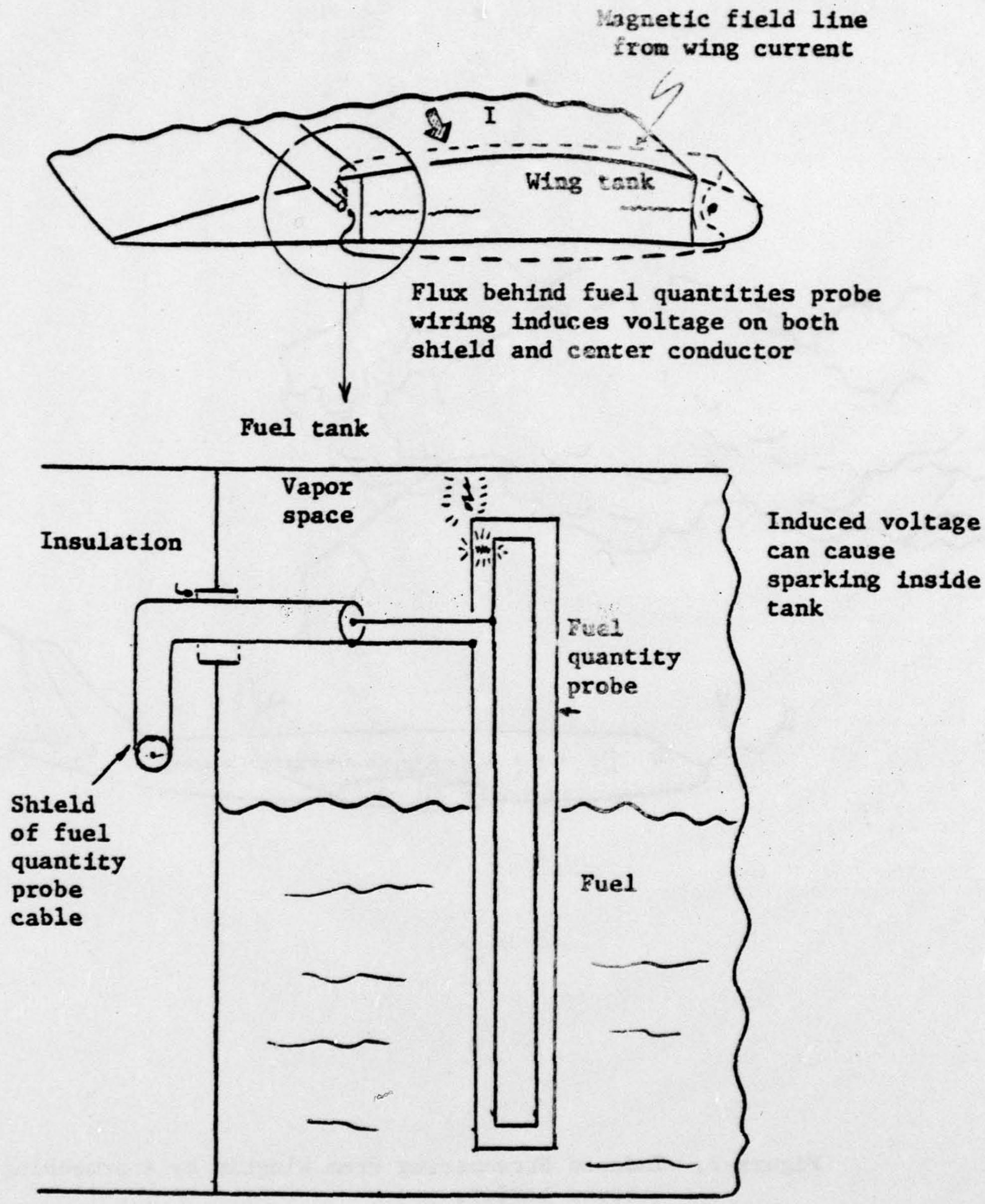


Figure 6. Fuel Quantity Probe Shield Can Pick Up Induced Voltages.

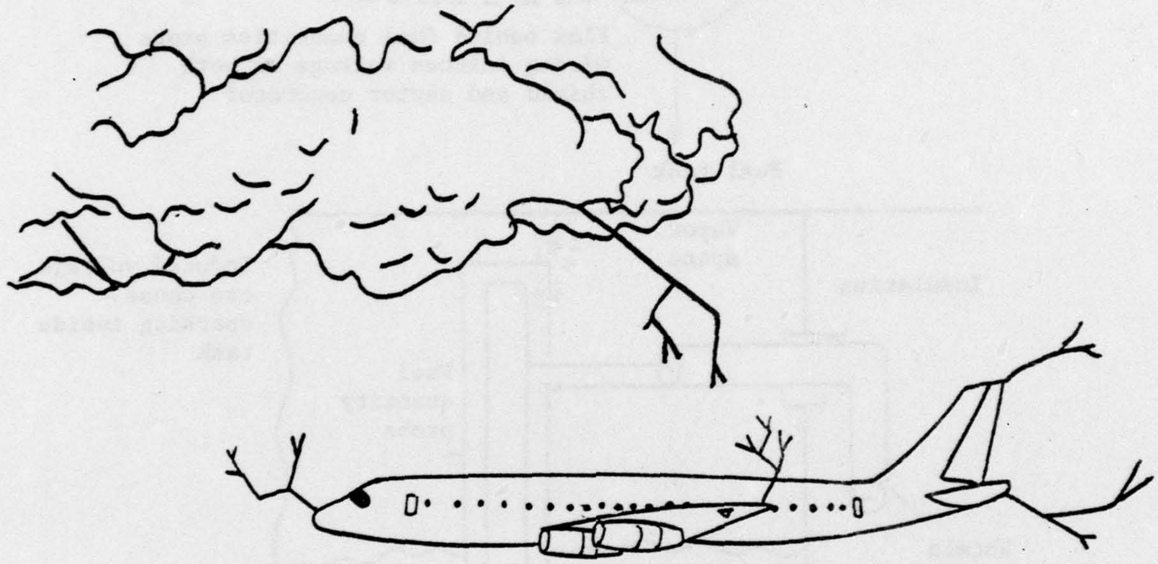


Figure 7. Induced Streamering from Wingtip by Approaching Stepped Leader.

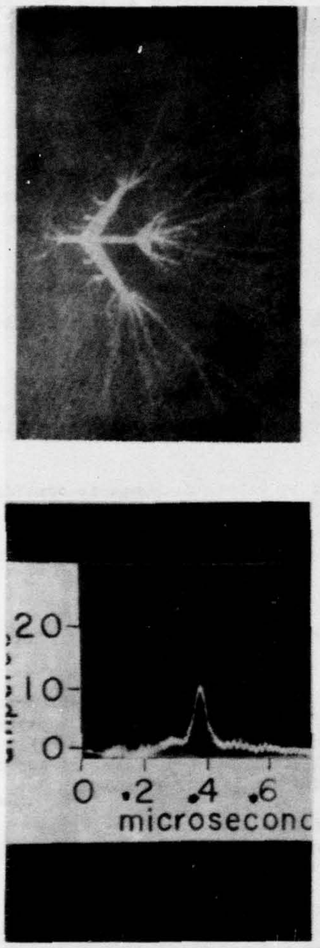


Figure 8a. Streamers on 1/50 scale aircraft model, and typical 12" length streamer branch current below, at 3 million volts.

Figure 8b. Streamers on full size NASA re-entrant wingtip vent, with absence of discharges on the fuel vent itself, lower photo, at 8 million volts over 60' to ground.

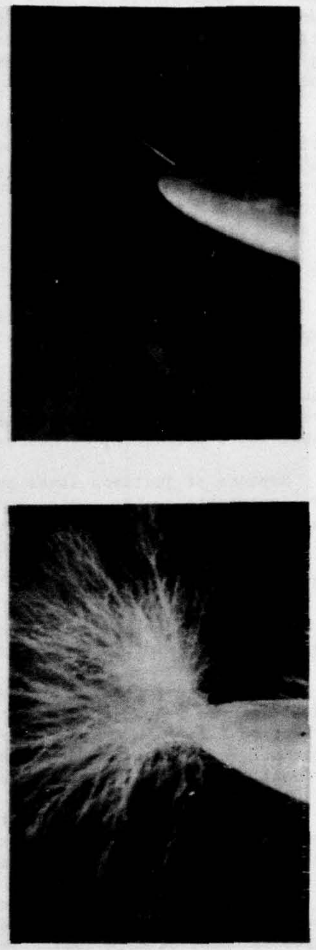


Figure 8c. Streamers from diverter are the most intense and would intercept and divert lightning strokes away from the upper edge shown in photo.

3.5 Flame Arresters

Considerable differences of opinion exist regarding the use of flame arresters on vent outlets. Some manufacturers use them for all vents and others manufacturers never use them and both have good justifications for their point of view. Flame arresters introduce the potential hazard of possibly icing and choking off the fuel vent and the fuel supply to the engines. However, it has been demonstrated that flame arresters can be used in fuel vents without icing problems when properly designed and located. The flame arresters in current use are generally of the egg crate type in which the flame arresting element has the depth of several inches to preclude the possibility of flame holding (heating of the flame arrester element and reignition on the fuel tank side of the flame arrester such as can occur with simple screen type flame arresters). Screen arresters can still be used in areas where direct lightning strikes are of a low probability as they will prevent streamering within the flammable vapor area of the tank and confine it to the exterior free airstream region where the ignition probability is extremely low.

Where arresters are located inside a vent line, the organ pipe resonance effects from external lightning strikes in the vicinity become of considerable importance. It has been shown that these resonant effects from the lightning stroke shock wave can result in the pumping of large quantities of combustion gas back and forth through the arrester at velocities which could possibly defeat the arrester operation and allow flame travel through. Thus the location of flame arresters at velocity nodes where vapor flow velocities are minimized is of importance.

3.6 Fuel Jettison Systems

Fuel jettison systems also require lightning protection and this is generally done by assuring that no complete open path exists between the jettison tube exterior and the fuel tank at any time.

This can be accomplished by use of gear type jettison pumps which block flame travel. This can also be done by arrangements to assure that liquid fuel completely fills the jettison line at all times such as fuel level switches which shut off valves on the jettison pump while liquid fuel remains over the jettison outlet. Flames don't propagate, of course, in liquid fuel.

Rupture of jettison lines by external ignition of vapor in tubes blocked by valves or pumps is considered very improbable as the combustion pressure is less than a few hundred atmospheres and in tubes a few inches in diameter, this results in hoop stress well below the elastic limits of aircraft aluminum. This is, of course, not true for larger aircraft sections such as wing boxes which can easily be exploded by ignited fuel vapors.

3.7 Fuel Tank Skin Treatment

Lightning protection for fuel tank skins involves (a) determination of the need of protection, if any and (b) the type of protection required.

Earlier researches determined the charge transfer magnitudes as a function of the arc time duration and skin thickness required for puncture and for hot spot ignition using stationary (non swept) arcs as discussed in Section 2.2. The mimimum energy and charge transfers occurred at arc durations of about 10 to 20 milliseconds. These earlier researches for unpainted skins had also shown typical stroke hang-on times of two to five milliseconds for titanium and aluminum respectively and even shorter times for stainless steel.

This data along with the earlier theoretical and experimental studies of time-temperature profiles (9) have indicated that without puncture, hot spot ignition with aluminum fuel tank skins is of very low probability but provided guidance on minimum skin thicknesses for unpainted skins.

More recent studies have shown that some of the newer skin materials such as stainless steel and titanium have larger time-temperature profiles and do present a potential hot spot ignition hazard, as illustrated in Figure 9. However more realistic tests with windstream cooling also show much reduced heating by any of the skin materials as illustrated in Figure 10. Although the preliminary hot spot investigations with windstream are encouraging, the area of hot spot ignition with windstream needs further research to define more precisely the safe envelopes of skin parameters for new materials.

Unpainted skins need less protection in general as the arc is swept by the windstream, moved laterally and longitudinally by the arc magnetic kink instability and is allowed to spread over a larger area where no paint confines it. When fuel tank skins are painted and lie in a Zone 2 area where arcs will be swept and are of marginal thickness, special paints are under development which help increase the number of arc contact points and reduce the energy in any single point.

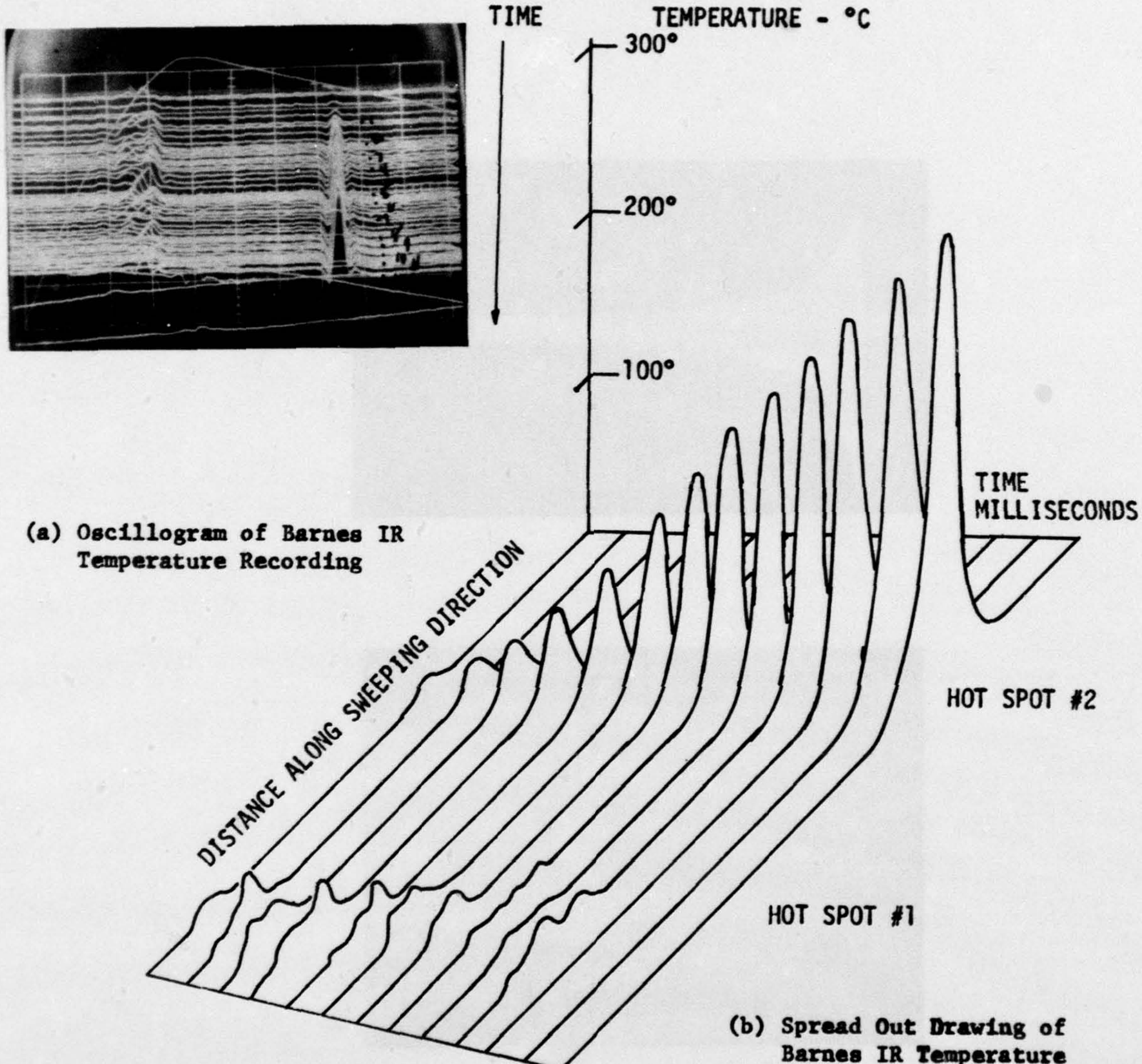
Insulation can be used (10) over metal skins in the local area to prevent arc contact. The insulation can also be combined with an external thin metal surface which acts as a sacrificial anode to move the arc as the metal burns away. This approach is used on several current jet transport aircraft.

For composite skins with either fiberglass or graphite fibers, external metallization can be used if required in the form of plasma flame spray, aluminum foil or screen mesh.

4.0 FUEL SYSTEMS TESTING

4.1 Fuel Systems Testing

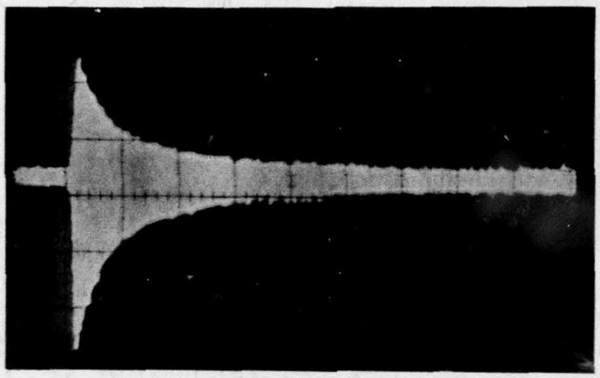
Of the three principal sources of fuel vapor ignition, the principal ones on which tests are usually carried out are fuel tank sparking and induced streamering and induced voltages on the fuel quantity probe wiring. Hot spot and arc puncture are taken care of by specifying skin thicknesses and surface treatments based on hot spot and puncture research data. They are seldom carried out for qualification tests as the test equipment and test procedures are expensive and time consuming. Hot spot testing requires infra-red imaging cameras which involves considerable time in making the measurements. In addition, wind stream is needed for realistic simulation and this further adds to the expense and time of the tests.

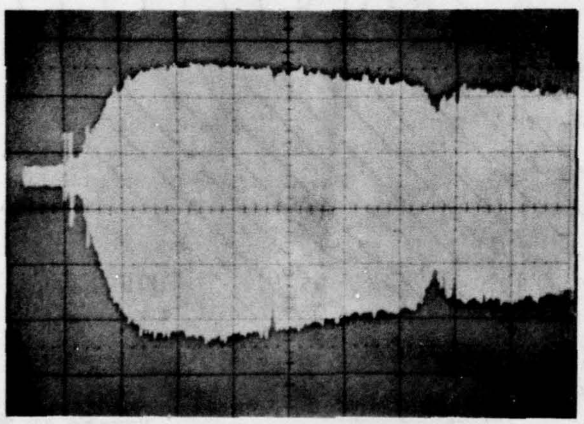


(a) Oscillogram of Barnes IR Temperature Recording

(b) Spread Out Drawing of Barnes IR Temperature Recording

Figure 9a. Multiple Trace Plots Illustrate Typical Barnes Infra Red Display with Ordinate Used to Display Swept Stroke Time Temperature History





HORIZONTAL SCALE - 0.125 SEC/DIV, VERTICAL - 314°C/ DIV

Figure 9b. Oscillograms of Peak Temperature Envelopes Show Great Variation in Rise and Decay of 0.060 Aluminum (above) and Stainless Steel (below).

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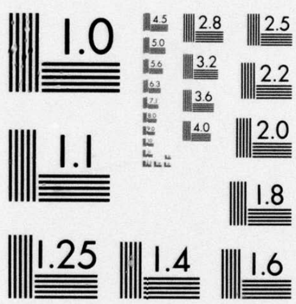

MICROCOPY RESOLUTION TEST CHART
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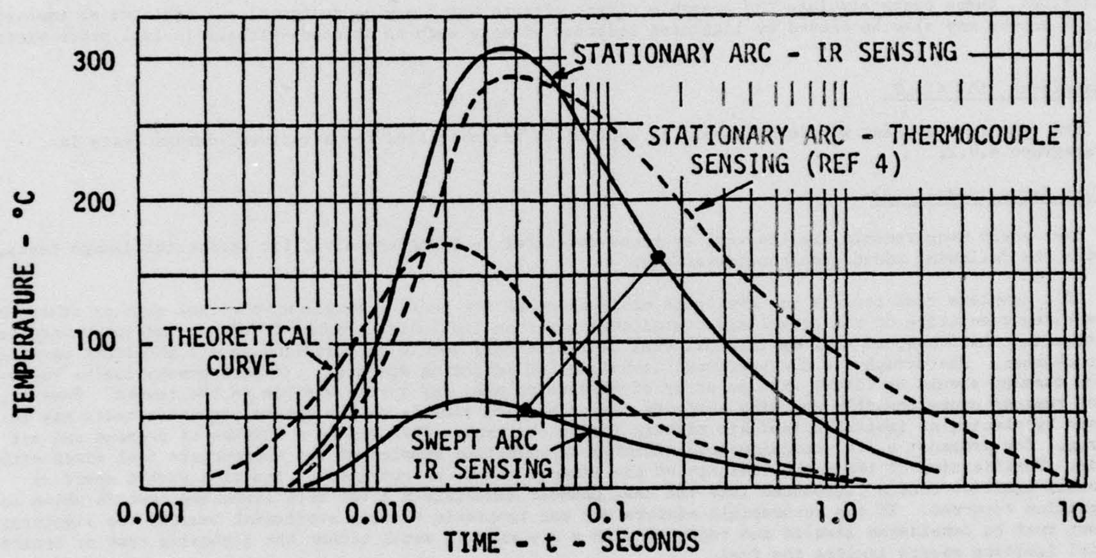

Figure 10. Measurements Show Greatly Reduced Hot Spot Temperatures with Swept Discharges.

Tests of induced voltage on fuel quantity probe wiring are similar to induced voltage measurements of other wiring which is discussed in other sections of the conference proceedings and are therefore discussed only briefly here.

4.1 Sparking and Streamering Tests

Sparking tests are preferably carried out using cameras to photograph any possible sparking or streamering. The tests are perhaps best described with reference to Figure 11 and to the present U.S. document on lightning testing by Subcommittee AE4 - Task F of the Society of Automotive Engineers Aerospace Council as excerpted below. (11)

EXCERPT FROM SAE AE4 - TASK F DOCUMENT, dated May 5, 1976:

Direct Effects - Combustible Vapor Ignition Via Skin or Component Puncture, Hot Spots or Arcing (SAE 4.5)

Objective (SAE 4.5.1)

The objective of these tests is to determine the possibility of combustible vapor ignition as a result of skin or component puncture, hot spot formation, or arcing in or near fuel systems or other regions where combustible vapors may exist.

CAUTION: These tests simulate the possible direct effects which may cause ignition. Ignition of combustible vapors may also be caused by lightning indirect effects such as induced voltages in fuel probe wiring, etc.

Waveforms (SAE 4.5.2)

The same test current waveforms should be applied as are specified for structural damage tests in Paragraph 4.4.2.

Test Setup (SAE 4.5.3)

Test setup requirements are the same as those described in Paragraph 4.4.3 for structural damage tests, with the following additional considerations:

If a complete fuel tank is not available or impractical for test, a sample of the tank skin or other specimen representative of the actual structural configuration (including joints, fasteners and substructures, attachment hardware, as well as internal fuel tank fixtures) should be installed on a light-tight opening or chamber. Photography is the preferred technique for detecting sparking. If photography can be employed, the chamber should be fitted with an array of mirrors to make any sparks visible to the camera. However, for regions where possible sparking activity cannot be made visible to the camera, ignition tests may be used by placing an ignitable fuel-air mixture inside the tank. This can be a mixture of propane and air (e.g., for propane: a 1.2 stoichiometric mixture) or vaporized samples of the appropriate fuel mixed with air. Verification of the combustibility of the mixture should be obtained by ignition with a spark or corona ignition source introduced into the test chamber immediately after each lightning test in which no ignition occurred. If the combustible mixture was not ignitable by this artificial source, the lightning test must be considered invalid and repeated with a new mixture until either the lightning test or artificial ignition source ignites the fuel.

Measurements and Data Requirements (SAE 4.5.4)

The same test current measurements should be made as are specified for structural damage tests in Paragraph 4.4.4.

The presence of an ignition source should be determined by photography of possible sparking. For this purpose a camera is placed in the test chamber and the shutter left open during the test. Experience indicates that ASA 3000 speed film exposed at F4.7 is satisfactory. All light to the chamber interior must be sealed off and the shutter opened to F4.7 aperture. Any light indications on the film due to internal sparking after test should be taken as an indication of sparking sufficient to ignite a combustible mixture.

CAUTION: This method of determining the possibility of sparking should be utilized only if all locations where sparking might exist are visible to the camera.

If a region where sparking may be possible cannot be made visible to the camera, ignition of a combustible mixture may also serve as a diagnostic 'measurement' to evaluate potential sources of ignition.

More specialized instrumentation may also be added if additional information such as skin surface tempatures, pressure rises, or flame front propagation velocities are desired.

Direct Effects - Streamers (SAE 4.6)

Objective (SAE 4.6.1)

Electrical streamers initiated by a high voltage field represent a possible ignition source for combustible vapors. The objective of this test is to determine if such streamers may be produced in regions where such vapors exist.

Waveform (SAE 4.6.2)

Test voltage waveform C should be applied for this test. The crest voltage should be sufficient to produce streamering, but not ~~sufficient to cause flashover~~ in the high-voltage gap. Generally, this will

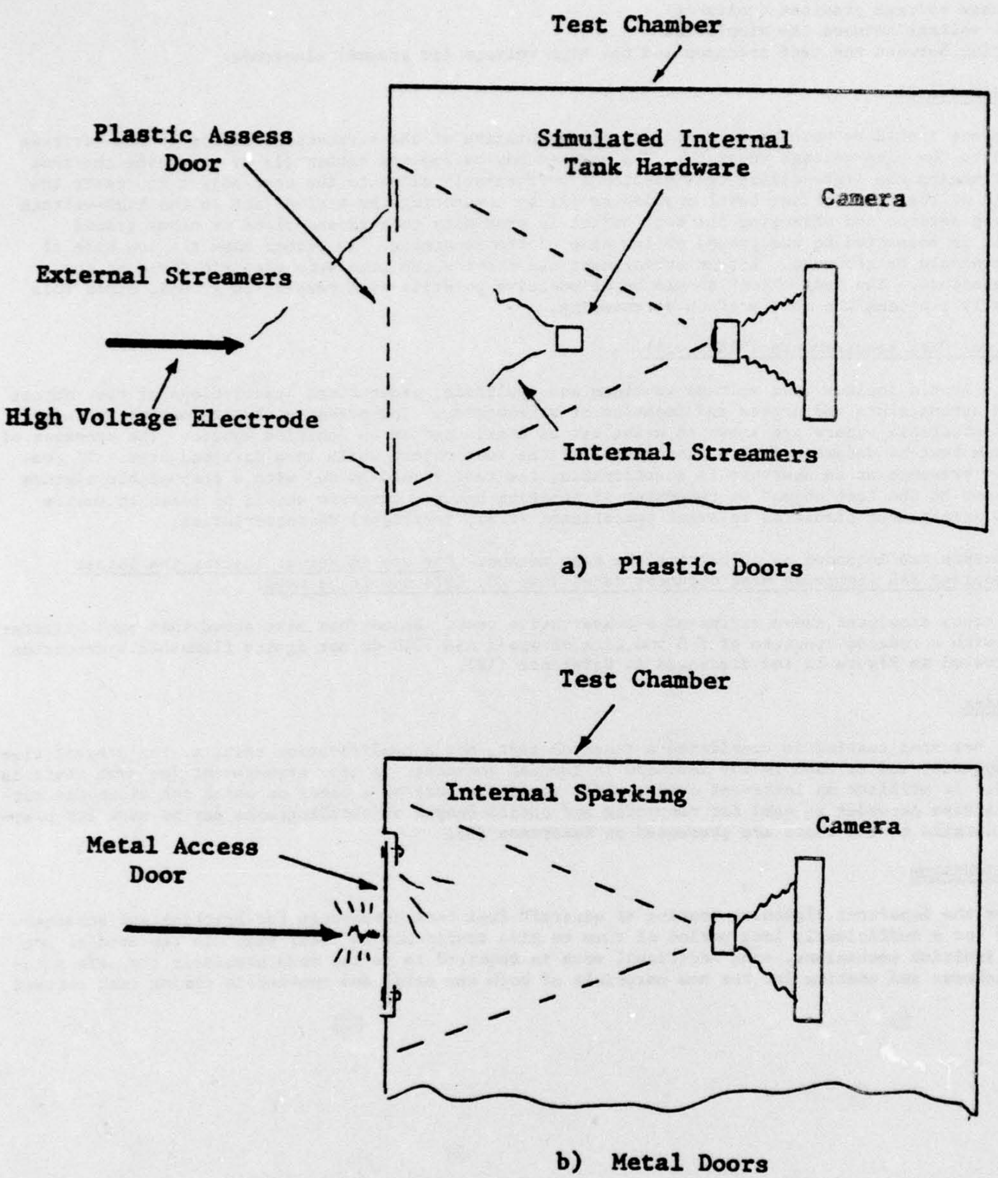

Figure 11. Test Arrangements For Fuel Tank Sparking And Streamer Testing.

require that the average electric field gradient, g_{av}, between the electrodes be on the order of 5 kV/cm, or

$$g_{av} = V/S$$

where

g_{av} = average voltage gradient (volts/cm)
V = peak voltage between the electrodes
S = spacing between the test specimen and the high voltage (or ground) electrode.

Test Setup (SAE 4.6.3)

The test object should be mounted in a fixture representative of the surrounding region of the airframe and subjected to the high-voltage waveform. The voltage may be applied either (1) by grounding the test object and arranging the high-voltage test electrode sufficiently close to the test object to create the required field at the test voltage level applied or (2) by connecting the test object to the high-voltage output of the generator and arranging the test object in proximity to a ground plane or other ground electrode that is connected to the ground or low side of the generator. In either case the low side of the generator should be grounded. Either arrangement can provide the necessary electric field at the test object aperture. The test object should be at positive polarity with respect to ground, since this polarity usually provides the most profuse streamering.

Measurements and Data Requirements (SAE 4.6.4)

Measurements should include test voltage waveform and amplitude, geometrical descriptions of test object and electrode arrangements and degree and location of streamering. The presence of streamering at locations where combustible vapors are known to exist may be considered as an ignition source. The presence of streamering can best be determined with photography of the test object while in a darkened area. If combustible vapor presence at an aperture is questionable, the test should be run with a combustible mixture actually present in the test object to determine if ignition occurs, but care should be taken to ensure that the test arrangement simulates relevant operational (i.e., in-flight) characteristics.

The above excerpts are intended to illustrate the test method. For use in actual testing the latest revision of the complete SAE lightning test document dated June 20, 1978 should be used.

The sparking tests excerpted above represent a conservative test. Researches have shown that much brighter sparks observable with a reduced aperture of F 8 and film of speed ASA 3000 do not ignite flammable hydrocarbon mixtures as illustrated in Figure 12 and discussed in Reference (12).

4.2 Hot Spot Testing

As discussed, hot spot testing is considered a research test, not a qualification test, at the present time because of the complexity and as such is not included in the SAE document. A test arrangement for such tests is shown in Figure 13. It utilizes an infra-red camera which scans the back of a panel on which the discharge currents are swept. A tape recorder is used for recording and oscilloscopes or oscillographs may be used for playback. Additional details of the tests are presented in Reference (9).

5.0 CONCLUDING DISCUSSION

Techniques for the important lightning testing of aircraft fuel tank components for sparking and streamering have been used for a sufficiently long period of time to give confidence in their use. In the area of hot spot and puncture ignition mechanisms, some additional work is required to define more precisely the safe parameters of skin thickness and coating for the new materials of both new metal and non-metals coming into current use.

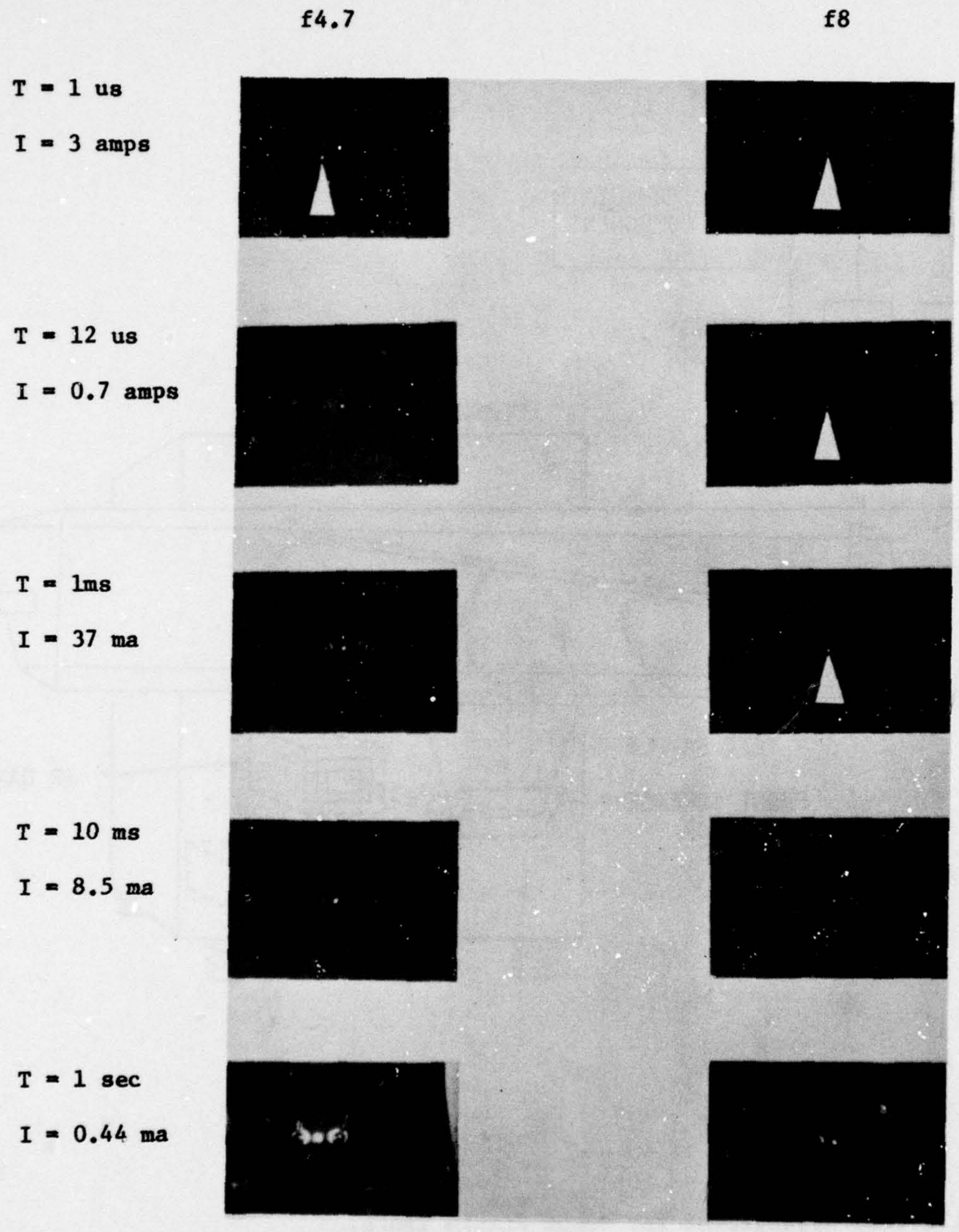

Figure 12. Photographic Records of Capacitor Discharge Sparks with Minimum Ignition Energy at f4.7 and ASA 3000 film. Even at f8, Sparks Are Still Visible.

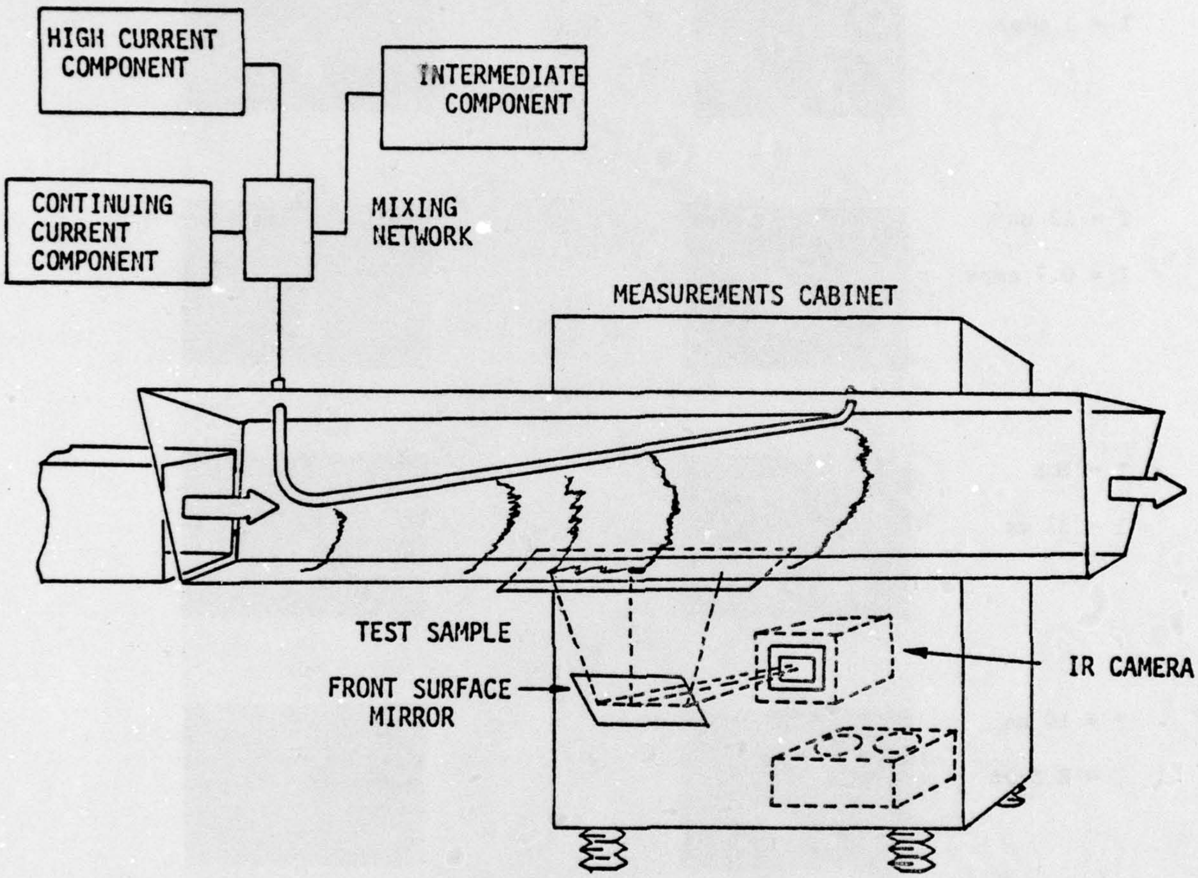

Figure 13. Test Arrangement for Swept Stroke Tests.

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QUESTIONS and ANSWERS

1 – From A.W. Bright to J. Robb and J. Plumer

Q – This question relates to the interpretation of data obtained with Test 303. When testing fuel tanks such as wing tip tanks, our own work on fuel to metal discharges inside tanks suggests that it may be difficult to photograph some very low energy discharges which have sufficient energy to ignite a fuel vapor. We have been studying discharges close to the minimum ignition energy and we find that we have to use image intensifiers to record the discharge. Also incidentally, we note that many discharges which produce a good deal of light are non-incendive. And the question is can one be certain that in the 303 tests on a fuel tank, a camera will record a potentially incendive discharge ?

A – LTRI tests of discrete sparks between electrodes with minimum ignition energies have shown that a camera with an aperture closed two stops less than that shown in Test 303 records all spark ignition, which suggests that the two stops smaller aperture indicates some margin of conservatism in the tests. This does not apply of course to brush type discharges such as Professor Bright would be photographing in fuel tanks under static electrification conditions. And it may not apply to brush-type streamer discharges inside a fuel tank from thunderstorm cross fields resulting from nearby lightning discharges. We would also agree with Professor Bright's comment that not all discharges which are visual may be incendive but this represents a conservative test for direct sparking. Professor Bright's comment that the image intensifiers are set at a level equivalent to that of the dark adapted eye corresponds to the sensitivity of the photographic test method of Test 303. This point certainly needs to be checked more carefully to determine quantitatively the possible levels of light intensity corresponding to incendiary sparks. The difficulty arises, of course, because of the wide variation in spark types from a high electron temperature glow discharges to arc discharges and molten metal particles. This is an extremely complex problem but we do feel that it certainly needs some serious efforts to provide more confidence in the tests, particularly for the streamering tests.

2 – From A.W. Hanson

Q – Experiments have been conducted at Culham to investigate the effect of lightning surge shock waves on flame propagation rates in fuel vapors.

Has LTRI done any work in this field and what were the results ?

A – LTRI has done some specific experimental investigations on lightning shock wave effects on flame propagation velocities as reported in FAA Report No DS-67-9 ("Airflow velocity Effects on Lightning Ignition of Aircraft Fuel Vent Efflux", Department of Transportation, Federal Aviation Administration, Aircraft Development Service, Washington, D.C. 20590, July, 1967). The experiments definitely showed high velocity flame propagation in the vents which resulted in high pressures and substantial vent distortion. It was suggested that the possibility of a detonation wave could exist when triggered by an initial shock which would not be possible under the quiescent ignition. Because of the ~~importance of the phenomena in relation~~ to flame detection and quenching systems we feel this needs further investigation.

BIOGRAPHICAL NOTE

J.D. ROBB

Attended Northwestern & University of Minnesota. After graduation worked for Honeywell & US Naval Research Laboratories. Then went to Lightning & Transients -presently Research Director.

Activities – Member IEEE & SAE-AE-4 Lightning Committee - Cochairman.

LABORATORY TESTS TO DETERMINE THE PHYSICAL DAMAGE (DIRECT EFFECTS) CAUSED BY LIGHTNING (QUALIFICATION TEST)

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SUMMARY

The paper describes the techniques that can be used to generate the test currents, conduct the tests, and to collect the experimental data for laboratory tests to determine the physical damage caused by lightning. Techniques for the simulation of the natural inflight strike environment, and the effects of arc length and other special factors are also discussed. Some guidance on instrumentation and diagnostics is also given.

1. Introduction

When an aircraft is struck by lightning the aircraft forms part of the total lightning path. Two attachment points are formed on the aircraft and lightning currents flow in the aircraft between these points. These currents can give rise to two types of effects defined as DIRECT EFFECTS and INDIRECT EFFECTS. This paper is concerned only with the DIRECT EFFECTS, which are defined in section 3.2 of the June 1978 draft of the proposals prepared by the SAE committee AE4L (1) as "burning, eroding, blasting, and structural deformation caused by lightning arc attachment, as well as the high pressure shock waves, and magnetic forces produced by the associated high currents".

The objective of this paper is to help towards a better understanding of the rationale of the tests normally employed, the criteria to be specified, and the interpretation of the test results.

2. Test Current Waveforms

The purpose of laboratory qualification testing is to assess the in-flight strike performance of an aircraft structure or component. The tests currents used should therefore effectively represent natural lightning in all aspects and parameters, and the test methods employed should effectively simulate the in-flight environmental conditions. Laboratory tests however are only simulations and the tests must be designed to enable reasonable allowances to be made in the interpretation of the test results for any shortfall in simulation of either the lightning currents or the in-flight environment.

Considerable work has been done collecting data of lightning strike parameters, and various tests waveforms have been defined which are intended to represent the various parameters of various phases of both positive and negative discharges. The waveforms of interest to physical damage direct effects tests are :- Current Components A,B,C, and D. These are defined in the proposals prepared by the SAE committee AE4L,(1) and in the UK Recommended Practice.(2)

The phase or phases of a lightning stroke, to which a particular component is likely to be subjected, will vary with the position of that component in the aircraft. In addition, each of the test current components have parameters giving rise to specific failure mechanisms. For example ohmic heating damage relates to $\int i^2 dt.$, while structural damage due to magnetic forces is a function of the peak current. The choice of the appropriate test current waveforms therefore must consider both the position of the test object on the aircraft, and the failure mechanisms under review.

It should be noted however, that where inductive sparking may be a hazard, eg. at fuel filler caps, currents having a high di/dt as well as a high peak value are required.

The rate of rise, and the rate of decay of the current pulse is also significant in tests for damage due to magnetic forces. The force generated is proportional to I^2 but the effective force will depend upon the relation between the angular frequency of the structure (ω), and the rise time and decay time of the current pulse. If the rise time $<< \frac{1}{\omega}$ and the decay time $>> \frac{1}{\omega}$ the effective force can be nearly doubled.

3. Generation of the Test Waveforms

The parameters of the components A,B,C and D, are very dissimilar, and in general different machines will be needed to generate them. These machines will generally be of the type where electrical energy is stored over relatively long periods and released in short pulses of high power levels. This may be achieved either by capacitive storage or inductive storage. The latter is the better simulation as the inductor acts like a fixed current generator, and in this respect is more comparable to natural lightning than a constant voltage generator, or a capacitative stored source.

CR 78 90

Nevertheless reliable and realistic testing can be effected with direct discharges from capacitor banks without the use of inductive storage.

With peak currents up to 200kA it is both difficult and expensive to charge an inductor from conventional d.c. sources. To overcome this capacitive storage can be used to charge the inductor. The simplest method of achieving this is to discharge the capacitor via the test piece into the inductor. At the first current peak all the remaining energy is stored in the inductor, and this can be fed into the test piece by simply short circuiting the capacitor. (see figs 1a, and 1b)

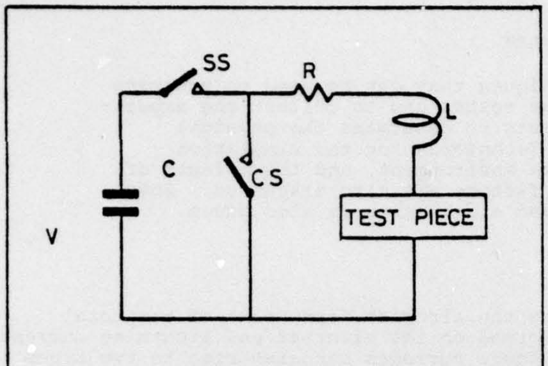

Fig. 1a

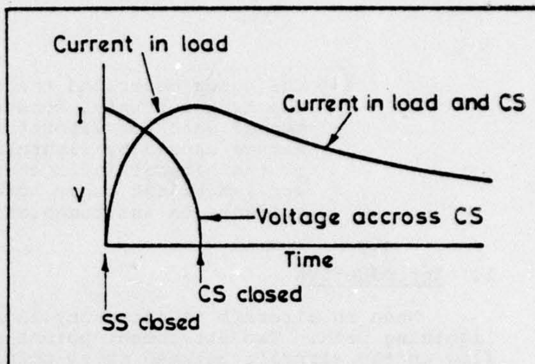

Fig. 1b

This system of initial capacitive storage discharged into inductive loads with a clamping or diverting switch, serves very well for simulation of the current component A, where the peak current, rise times and pulse duration permit the design of a generator having practical dimensions. The main limitation of this system however is the control of the rise time, which is a function of $\sqrt{L,C}$. Where high charge transfers and medium currents are required, (such as in current components B and C) both C and L need to be large; and this gives a very long rise time for current in the inductor. Faster rise times can be achieved by discharging the capacitor directly into the inductor, and discharging the inductor into the test piece after the transfer of stored energy form capacitor to inductor has been completed.

Additional equipment can be used for the current component C. Theses include DC generators and battery banks. Such devices however are suually constant voltage sources and are not, in general, as good a simulation as an inductive storage system. This is particularly true of open arc work where a constant voltage source puts constraints on the arc behaviour not representative of the natural lightning condition.

The type and nature of the test current generators is one of the factors that the certification authorities should take into account in assessing the results of certification tests.

4. High Current Measurement and Recording Instruments

The two basic measurements required are current and voltage. Current measurements can be made by two basic methods:

(a) Low inductance resistive shunts.

(b) Magnetic probes, pick-up coils, or Rogowski coils.

Shunts are most useful for the intermediate current and continuing current phases, where the di/dt is low, the duration relatively long, and the circuit can tolerate the insertion inductance. Current peaks vary from 100 A or so to some tens of kiloamperes. Calibration is absolute, and the diagnostic is robust, consistent, and reliable. The shunt must be able to carry the $\int i^2\,dt$ of the high current circuit, and it must be introduced at the earth point only. Care must be taken not to introduce earth loops.

Rogowski coils and magnetic probes may be used for the higher peak value currents (say $>10^4$A) and di/dt values (say >5 kA μs^{-1}). The Rogowski coils have the advantage of absolute calibration but their high frequency performance is poor. These two sensors have no metallic connection with the high current circuit and therefore they do not require reference to the earth point. They are however more susceptible to high frequency pick up and hash originating from the very high peak current trigger pulses and other switching noises.

The elimination of hash can be greatly facilitated by the use of balanced twin cables, in solid copper tubes. The copper tube must be kept directly against, and in electrical contact with the high current transmission line at its lowest field point, from the position of the diagnostic to the common earth point.

Other current measuring devices based upon magnetic field measurement such as Hall effect probes could be used. In general however they have no advantages over the simple coil magnetic probe.

Voltage measurements can conveniently be arranged with suitabel potential dividers.

Here again the earth reference point must be connected directly to the low voltage end of the potential divider, and to the earthy leg and screen of the balanced twin cable. Earth loops must also be avoided. Analysis of these two functions (ie. voltage and current) with respect to each other can give a very good indication of the performance of the test specimen.

In open arc work, high speed cine films should be regularly employed. Still photography and light sensitive detectors can also add useful information to the diagnostics. Pressure sensors, accelerometers, and heat detectors also have an important role to play for appropriate tests.

5. Simulation of the Natural Environment

Simulation of the natural environment is an extremely important part of lightning testing, as incorrect simulation may give misleading results.

5.1 Simulation of the Lightning Current Paths

In the natural environment the charge is simply transferred from one cloud area to another, or from cloud to ground. The aircraft is therefore in free space with no return path current magnetic fields to influence the behaviour of the lightning currents flowing through it. In laboratory conditions return path conductors are always associated with the test piece, and their associated magnetic fields can influence the test results. In tests on components in zones 1 and 2 where open arcs are used, the arc path can be severely distorted by magnetic forces, and in zone 3 tests the path taken by the current through the test object can be influenced by the position of the return current conductors.

One solution to this problem is to have multi-return paths so arranged that the individual magnetic fields summate to zero in the region of the test object. In most cases this can be achieved with 3 or 4 return conductors. In some cases only two wide transmission plates may be nescessary. With very larger objects, and particularly for currents with high di/dt an assessment must be made of where the currents will flow, and how they will redistribute themselves during the pulse. The return conductor geometry may then be designed to reproduce that effect. This final design will usually be a compromise between the load inductance requirements, and the environment simulation requirements.

5.2 Arc Length and the Arc Root

In early work on arc root burn through of metal panels, the results were strongly influenced by the arc length. This was found to result from the presence of electrode jets. These jets are emitted from arc roots on both electrodes and consist of a jet of ions, neutral particles and clusters of neutral particles. High speed cine film shows that these jets are very active for up to 50mm from each electrode, and have a strong inter-action with each other. In a natural strike to an aircraft, only the jet associated with the attachment point to the aircraft itself exists, and there is no jet corresponding to that emanting from the test electrode in a laboratory simulation. It is therefore desirable to separate the electrode jet from the current channel and direct it away from the test object. Since the electrode jet is always emitted normal to the surface of the electrode at the arc root, the jet may be directed from the test object by redirecting the arc root to a suitable angled facet of the electrode by means of a suitable insulator.

Experiments with this type of electrode have given results which are sensibly independent of arc length indicating that the electrode jet effect has been virtually eliminated. Where this type of electrode is not used much longer gap distances are required to avoid the effects of jet interaction. Ideally a distance of about 150mm is required to ensure no jet interaction, but practical difficulties of arc stabilization, and driving voltage forces some compromise. The SAE Proposals (1) recommend 50mm for current components A and D and 10mm for combined waveforms. This reduced length of 10mm is in general a more severe test.

5.3 The Effects of Forward Speed

The forward speed of the aircraft causes the lightning attachment points to sweep rearwards in discrete steps. For arc burn through tests in the swept stroke region it is useful to know the maximum arc dwell times. Techniques for establishing this have already been described at this conference by D. Clifford (US) and P Little (UK). Where it has not been possible to firmly establish a maximum possible dwell time however, a dwell time of 50ms should be assumed.

Experience has shown that when a restrike occurs in a swept stroke region, a new attachment point is always formed, so tests for Zone 2a* therefore have the current components applied in order of :- D(representing the restrike) followed by B and C (representing the subsequent intermediate and continuing currents). Tests for Zone 1A require only current components A and B as the attachment point will have swept backwards before the event of the continuing current (Component C) or the restrike (Component D). Zone 1B is an initial attachment which cannot sweep anywhere else and will therefore see all current components of the strike in the order in which they occur viz A.B.C and D. Zone 2B. however does not see the initial return stroke but could see

all the remaining components in the order in which they occur viz B,C and D. Components in zone 3 can also see all the components of the lightning strike in the order in which they occur. There is however no arc root problem here so it is only normally necessary to test for the high peak current and high action integral of current component A, and the high charge transfer of current component C. Where inductive sparking may be a hazard however current component A should be applied with the appropriate di/dt, or alternatively current component E should be applied.

* See table on Page 4 for definition of lightning attachment zones.

5.4 Other Environmental Parameters

Other factors of environment such as temperature, and atmospheric pressure are in general less significant. In some special cases however, eg. fuel ignition, they may play a significant part, and must not therefore be entirely ignored. Simulation of these parameters can be difficult, and need only be attempted where there is very good reason to do so.

6. Special Hazards

Extra care should be taken where special hazards exist. These could take the form of unusual materials or particular construction methods. Examples of these are carbon fibre panels, where surface damage and heat damage may be high; thin metal skins where arc burn through is probable; fine wires encapsulated in sinsulators (and the use of aluminium honeycomb sandwich panels) where explosive fusing is possible; and glued joints where current may cross the glue line causing delamination of the glued joint.

7. Interpretation of Results

It is important to have more than one diagnostic device, preferably of a different type, so that the results from one diagnostic can be cross referenced and checked by the results from the second diagnostic. In open arc work a good indication of the arc behaviour during the pulse can be obtained by comparisions between the arc voltage and arc current traces. With inductive storage generators the current decay characteristics are determined by the discharge circuit characteristics eg. resistive or arc. Thus a mid-pulse change in circuit characteristics such as the development or quenching of an arc, can be precisely determined by analysis of the current decay characteristics.

When ever the results are indicated by one diagnostic instrument only, efforts must be made to produce a working theoretical model based on any deductions made from the results. When the results were both expected and hoped for it is all too easy to overlook the possibility that such encouraging results were merely due to fortuitously chosen test conditions. It is equally hard to avoid the temptation to make theoretical extrapolations indicating possibilities of success that may well be illusory in the wider field. Finally there is the all important post-test inspections and measurements, and the comparison with the pre-test inspections. Clearly however, the success of this all important operation depends largely upon how thoroughly the pre-test inspections were conducted.

Acknowledgements

The author acknowledges the support given by the Ministry of Defence for the work conducted at Culham Laboratories.

Definition of Lightning Attachment Zones

Zone 1	Surfaces of the aircraft for which there is a high probability of initial lightning flash attachment.
Zone 2	Surfaces across which there is a high probability of a lightning flash being swept from a zone 1 point of initial flash attachment.
Zone 3	Zone 3 includes all of the aeroplane surface areas other than those covered by zones 1 and 2.

Zones 1 and 2 may be further subdivided into A and B regions.

Zone 1A	Initial attachment point with low probability of flash hang on, such as a leading edge.
Zone 1B	Initial attachment point with high probability of flash hang on, such as a trailing edge.
Zone 2A	A swept stroke zone with low probability of flash hang on, such as wing mid-cord.
Zone 2B	A swept stroke zone with high probability of flash hang on, such as a wing inboard trailing edge.

References: *1. Lightning and Qualification Test Techniques for Aero Space Vehicles and Hardware Draft Proposals by SAE Committee AE4L. 20 June 1978*
2. Recommended Practice for Lightning Simulation and Testing Techniques for Aircraft. H.M. Stationery Office Report CLM-R163.

QUESTIONS and ANSWERS

– From S.D. Schneider

Q – Does the use of your dielectric controlled discharge probe (Jet diverting electrode) create arc wander and therefore less repeatable data ?

A – No ; on the contrary the use of this electrode produces more repeatable results. In the experiments described stray magnetic fi   s were eliminated, and this is one of the major causes of arc wander ; and the elimination of the test electrode jet removes the interaction between the two arc jets thus reducing the arc path instabilities that arise from such interactions. In some cases new attachment points can be created on the test sample by impingements of the test electrode jet. This problem is also eliminated with the removal of the test electrode jet.

Laboratory Tests to Determine the Physical Damage (Direct Effects) Caused by Lightning

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Summary

The relatively high costs of lightning investigations for aircraft or missile components are justified by the fact that by far higher valuables may be lost by the damage or loss of an aircraft or missile struck by lightning. Besides the loss of material, such as an aircraft, the human life must always be given the primary and highest rate.

The use of new fiber material, for example for air vehicle engineering, made extensive investigations of lightning protection systems necessary. Based on these considerations a test facility for simulating lightning has been established at the Plasma Institute of the Technical University of Hannover for test series covering especially CFC-samples. When designing the facility it has deliberately been kept within narrow limits in order to keep the costs low one hand and to be able to start with the tests as soon as possible on the other hand. Therefore, only direct effects were investigated up to date on CFC-samples equipped with different lightning protection systems.

The specific function of the test set-up at the TU-Hannover is to simulate lightnings with the both characteristics, the initial stroke with a peak amplitude of 200 kiloamperes in a period of 12 microseconds and a subsequent continuing current with a charge transfer from 0 to 500 coulombs over a period of 0,5 seconds. For our purpose the tests were performed at the following test conditions:

200 kiloamperes	/ 0 coulomb		
200 kiloamperes	/ 50 coulombs	/ 0,5 seconds	
200 kiloamperes	/ 210 coulombs	/ 0,5 seconds.	

According to our opinion this covers a wide range of lightning simulation also for the design techniques for advanced composite structures.

The sample size was defined to be 400 x 400 millimetres, the thickness of the plates only a few millimetres.

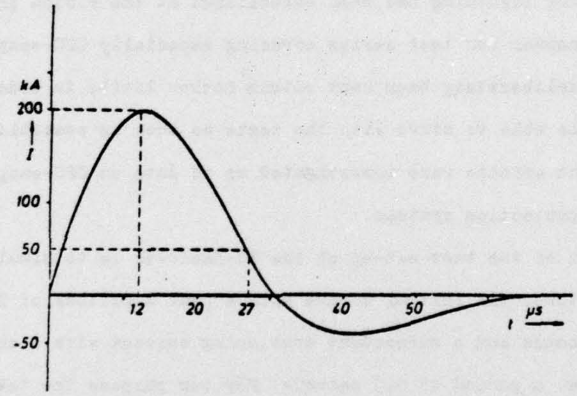
Prior to the actual investigations a few "test shots" were made to determine the optimum electrode spacing, sample support mechanism, cables and capacitance of the lightning simulation system.

Test Facility

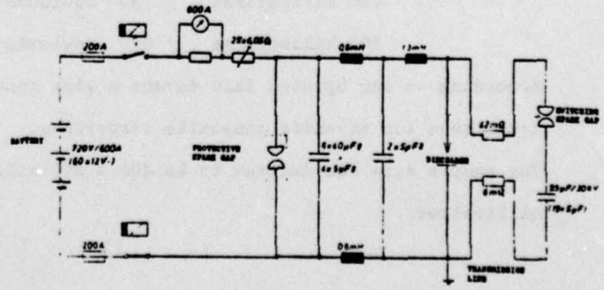
According to the findings of lightning protection research a lightning consists of two phases. The first phase with a duration of only a few microseconds includes a very high surge voltage and surge current load, respectively, with a minor charge transfer. During this phase a conductive spark channel is built up permitting a high current flow with a charge of several hundred coulombs via the spark plasma during the subsequent second phase of several orders longer duration. In addition to these two basic phases of a lightning one or more high spike-like "restrokes" may occur during the high-charge phase.

For simulating lightnings a facility was established in the laboratory of the Plasma Institute of the Technical University of Hannover permitting the simulation of the first two phases of a lightning subsequently or independently from each other.

Initial stroke. The initial stroke is generated by the facility as a pulse with a rise to 200 kiloamperes during a period of 12 microseconds. The decay to 50 kiloamperes takes place within the next 15 microseconds (figure 1).



The pulse is generated by a bank of capacitors with a maximum capacity of 95 microfarad selectable in steps of 5 microfarad (19 capacitors of 5 micorfarad each). (Figure 2).



The maximum charging voltage is 40 kilovolts, the voltage normally used for the test was 30 kilovolts.

The energy content of the initial stroke at a charging voltage of 30 kilovolts amounts to 42.75 kilojoule.

Since a voltage of 30 kilovolts is present between the specimen and the backplate electrode at the point of ignition, the bank must be protected against this high voltage pulse. As protective device a two-stage filter system is used representing a high impedance for the frequencies to be expected. For the case that no spark-over occurs between specimen and backplate electrode the energy stored in the bank of capacitors is dissipated on a low pressure protection spark path with an adjustment threshold voltage of 900 volt located behind the filter stage.

The spark-over to the specimen caused by the initial stroke closes a second circuit in which a current of maximum 750 amperes is made available for a period of 10 seconds at a conducting voltage of 720 volts from 60 starter batteries. Dropping resistors and a timing circuit are provided for the pre-selection of the current and the load duration. The normal values for the tests were 100 to 500 amperes and 0,5 seconds.

Energy is supplied to the specimen by means of an extremely low-inductive steel strip of a length of 2,75 metres. The lower conductor consists of U-iron (50 millimetres large) and has an effective ohmic resistance of 6 milliohms. The upper conductor is a 0,5 millimetres thick steel strip (40 millimetres large) with an effective resistance of 42 milliohms.

The transmission line serves as loss resistance for the almost aperiodically attenuated cycle of the initial stroke, it terminates in the electrode support and an adapter to which the specimen is fitted.

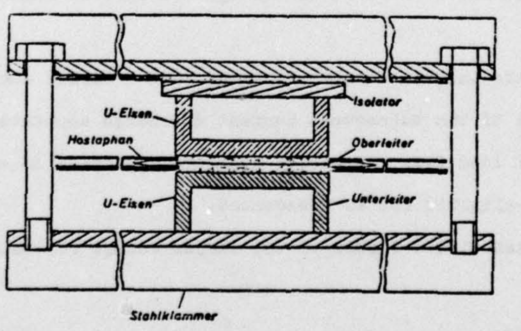


Abb. 3: Querschnitt durch das Bandleitersystem

A semicular round tungsten rod with a diameter of 10 millimetres is used as electrode.

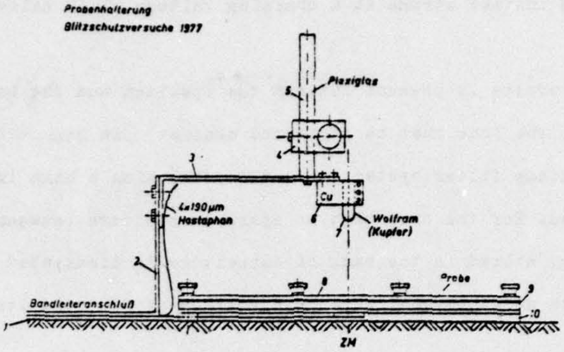


Abb. 4: Probenhalterung für 400 × 400 mm² große Platten

The surface of the electrode is always re-finished depending on the burnoff to ensure a constant transfer resistance.

During the initial stroke the current is measured by means of Rogowski coil, the signal of which is electronically integrated and, therefore, proportional to the current. The Rogowski coil with the integrating system has been calibrated by means of a high-current tuned circuit, the current flow of which can be determined very precisely from the measured circuit data (resistance, capacitance, inductivity and voltage).

During the high-coulomb discharge the voltage drop across a dropping resistor is used for measuring the current. The voltage is measured at the specimen by means of high-voltage probes according to the differential measuing method.

For all parameters the temporal flow is recorded in a screened measuring cabin with 10 megahertz oscilloscopes and recorded on film.

For a load example picked out at random the voltage and current signals are represented in Annex 16-A-1.

Voltage and current of the initial stroke, in Annex 16-A-1 called surge current, are recorded together and the current of the subsequent current discha[illegible] seperately in this case. The deviations of the actual load from the design load are dependent on each discharge, but they are insignificant and negligible for an assessment.

Annex 16-A-2 shows the associated figure of the sample damage for this special load case.

Conclusions and Prospects

The investigations made so far produced well usuable results. It turned out that with the established simulation facility representative, i.e. well reproducible, results can be obtained under appropriate test conditions. A comparison was drawn in France under almost the same tes conditions.

Measurements resulted in a good agreement of the figures of damage.

Further investigations with selected samples will follow, an extension of the test series is planned.

References

Entwicklung von Blitzschutzsystemen für Bauteile an faserverstärkten Werkstoffen, ZTL 1977 - 78; FAG-2.

Dr. Th. Reese,

Versuchsanlage zur Simulation von Blitzeinschlägen; Institut für Plasmaphysik, TU-Hannover.

Appendices

16-A-1 voltage and current

16-A-2 sample damage picture

16-A-1

Test Number 43 System Number III

Current Load

Nominal Actual

200 KA 220 KA

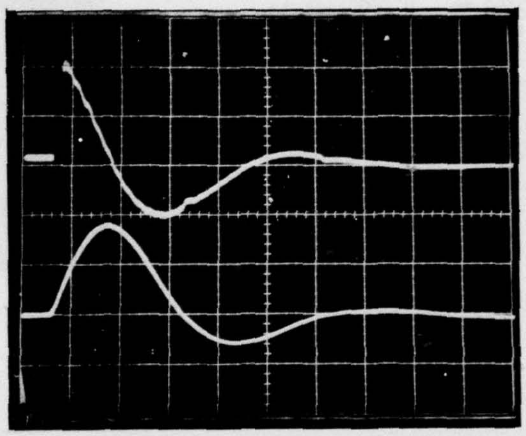
210 C/0,5 s. 239C/0,54 s.

Voltage Signal: 5 KV/SKT

Current Signal: 125 KA/SKT

Time Axis: 10 /us/SKT

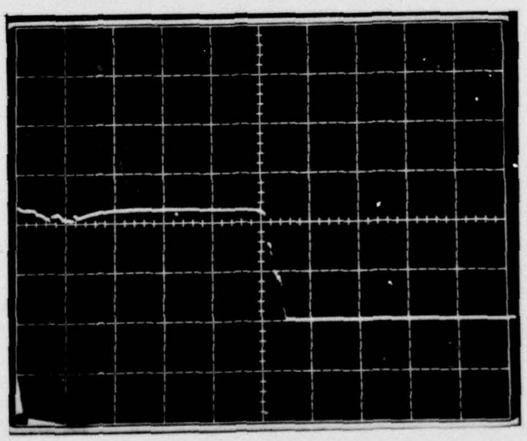
Surge Current Discharge



Current Signal: 196 A/SKT

Time Axis: 0,1 s/SKT

Subsequent Current Discharge



Test Number 43 System Number III

Notes: 220 KA, 239 C, 0,54 sec.

Sample Face:

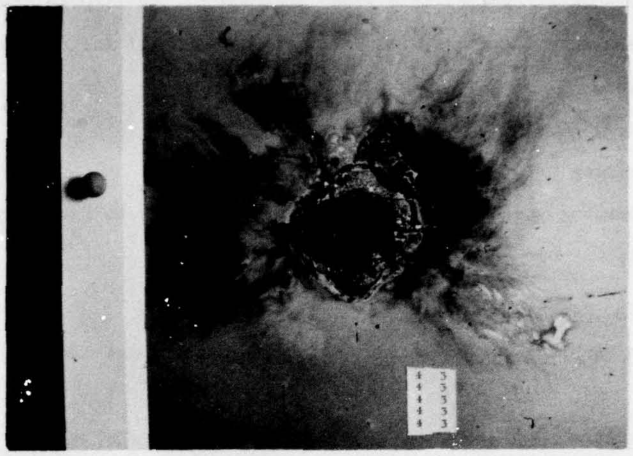
SS: NLP 55 mmØ, peeled off

OS: ALF 50 55 mmØ, molten

RS: 319 A 50/40 mm burned

US: ALF 25 50/40 mm molten

CFK 10 mmØ, hole



Test Number 43 System Number III

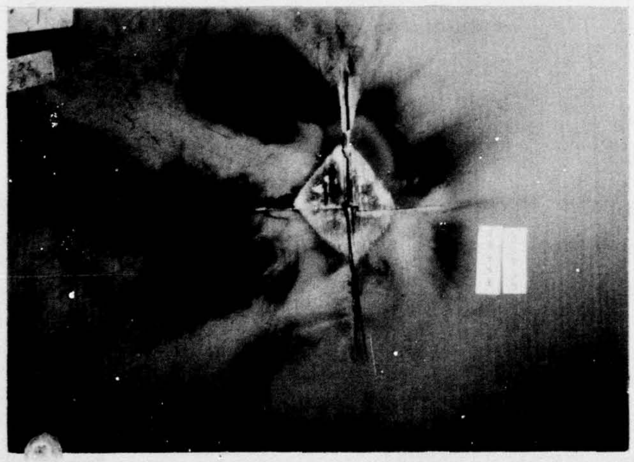
Notes: 220 KA, 239 C, 0,54 sec.

Back of Sample:

50 mmØ burned area

120/80 mm $0^\circ/90^\circ$ cracks

10 mmØ hole



QUESTIONS and ANSWERS

Q – What is the $\int i^2\,dt$ of your high current pulses ?

A – Action Integral $\int i^2dt(A^2s)$ of Initial Stroke

The surge current consists of the first positive half-wave and the subsequent negative half-wave.

The peak amplitude of 200 kA ± 10% is reached in a period of 12 μsec, the zero-axis crossing of the first half-wave takes place after 30 μsec.

The negative half-wave crosses the zero-axis after approx. 65 μsec. The total charge of the bank of capacitors is Q = 2.85 As at a charging voltage of 30 kV ± 10%.

For an actual measured surge current with a peak amplitude of 210 kA at 28 kV the following $\int idt$- and $\int i^2dt$-values have been determined for the first and second half-wave, respectively.

1st half-wave : $\int_{I} idt = 3.84$ As

2nd half-wave : $\int_{II} idt = \underline{1.55 \text{ As}}$

$Q_{total} = 2.29$ As

At 28 kV the total charge of the bank of capacitors equals

$$Q_{total} = 95.10^{-6} \times 28.10^{3} = 2.66 \text{ As}$$

The deviation from the value determined from the oscillogram is caused by the fact that the third half-wave has not been considered on the one hand and that charge leakages occur on the other hand.

For determining the action integral $\int i^2dt$ the current is squared point by point.

The measurement of the area resulted in the following values :

1st half-wave : $\int_{I} i^2dt = 580 \times 10^3\ A^2s$

2nd half-wave : $\int_{II} i^2dt = 140 \times 140 \times 10^3\ A^2s$

$\int_{I} i^2dt + \int_{II} i^2dt = 720 \times 10^3\ A^2s$

The action integral in this case will therefore be :

$\int i^2dt = 0.72 \times 10^6\ A^2s.$

BIOGRAPHICAL NOTE

J. SKIBA

Test Center, Manching (Germany)

University of Berlin. Graduated as Dipl. Ing. (master). 1960-67 Systems engineer for Cathodic Protection. Since 1967, Civil Servant with German MoD. 1969/70, 1 year for exchange at Westinghouse in Lima Ohio (U.S.). 1970-1978 at Flight Test Center in Manching. 1976 1 year detachment to MoD Bonn.

Responsible for Aircraft Electrical Systems, Anti Icing Systems, Lightning Protection and Static Discharging.

Married since 1960 - two children.

LABORATORY ~~TESTS TO SIMULATE~~ LIGHTNING STREAMERS AT APERTURES (A QUALIFICATION TEST)

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SUMMARY

When an aircraft becomes subjected to an electric field of sufficient intensity, corona and streamering may occur at locations where the electric field exceeds the corona inception level. Often these discharges occur at extremities where the radii of curvature is small, such as pitot probes, antennas, and wing, empennage or propeller tips; but they may also appear at discontinuities such as windshields or fuel vent outlets, or beneath dielectric covers. Since these discharges may have sufficient energy to ignite fuel vapors, it is frequently necessary to determine if they may occur within non-metallic fuel tanks and other enclosures that may contain such vapors. Streamers may also induce severe electrical transients in electrical circuits feeding windshield heaters or other exposed systems. A test in which the electric field that produces these streamers is applied to the aircraft has been included in the new standard. Methods of performing this test are described.

Origin of Corona and Streamering

An aircraft in flight may become electrically charged to relatively high voltage potential by one or more of several mechanisms. These are:

- Direct lightning strike attachment.
- Presence in an ambient static electric field (sometimes called a "cross-field").
- Triboelectric charging during flight through areas of precipitation or other particulate matter.

These voltages may equal 500 kV or more, creating an intense electric field about the aircraft. This field is sufficient to produce a violet-colored luminous ionization, called corona, around wing tips, propeller blades, trailing edges and other sharp appendages. This corona is often referred to as St. Elmo's fire. Corona occurs at a conductor when the electric field surrounding it reaches the ionization potential of air, which is about 30 kilovolts per centimeter at sea level and somewhat less at flight altitudes. These fields may be either impulsive, as in the lightning-strike case, or constant (DC) as in a cross-field or precipitation charging situation. In either case, the voltage level which the aircraft must be at with respect to its surroundings when corona begins to occur is called the *corona inception level*. This depends on the geometry of the aircraft and, in the laboratory, upon the proximity of an oppositely charged electrode, and is typically about 100 kilovolts.

Objects with sharp radii of curvatures intensify the surrounding electric fields and produce corona at lower inception voltages than rounder, smoother surfaces. Corona is accompanied by a liberation of heat, and can liberate sufficient energy to cause ignition of a flammable vapor. Corona is also accompanied by broadband electromagnetic radiation which can produce interference in susceptible aircraft avionics and communication systems.

When the aircraft voltage level increases beyond the corona inception level, discrete discharges may propagate out from the surface, appearing like miniature lightning flashes. These are sometimes called "static discharges". The current levels are between several milliamperes and a few amperes, and as such are not capable of damaging metallic aircraft structural materials. In some cases, static discharges have punctured windshields or fiberglass skins, but no significant damage has been reported from such discharges. Direct lightning strikes, in contrast, can inject damaging currents into the aircraft.

Laboratory Simulation

Simulation of corona and streamering in the laboratory is a relatively simple affair; it involves subjecting the test object to an electric field capable of producing corona and streamering, and detecting the location of these effects. Whereas corona and streamers may be produced by either DC or impulse fields of very short duration, it is often most practical to apply an impulsive field of intensity sufficient to produce corona and streamers but of insufficient duration to allow complete flashover of the electrode gap. Such a test has been included in the new standard (Reference 1), and is described in that reference as follows:

"TEST METHOD 303

CORONA AND STREAMERING

1.0 PURPOSE

This method is used to determine if electrical streamers or corona can be produced at or near apertures or at other locations where corona and streamering may be of concern.

2.0 APPLICABILITY

This test is applicable to fuel vent and dump outlets, radomes, antennas, canopies and other components which are exposed to thunderstorm electric fields.

3.0 APPARATUS

The test apparatus shall include:

a. A high voltage generator capable of producing voltage waveform B with a crest voltage sufficient to produce the electric field specified in Para. 5.

b. A test electrode of a configuration that would produce a uniform electric field over the general contour of the test object.

c. High voltage measuring and recording equipment.

d. Photographic equipment capable of detecting corona and streamers, and a suitable environment for such photography.

4.0 TEST SETUP

The test object shall be a production-line hardware component, a full-scale replica, or a section of the aircraft structure, mounted in a fixture representative of the surrounding region of the airframe. The voltage may be applied either by (a) grounding the test object and locating the high voltage test electrode sufficiently close to the test object to produce the required field at the test voltage level applied or (b) connecting the test object to the high voltage output of the generator and locating the test object in proximity to a ground plane or other electrode that is connected to the ground or low side of the generator. The test object shall be at positive polarity.

If all possible internal streamer sources cannot be observed by the camera or by the camera plus mirrors, the test shall be run with a combustible vapor inside the structure to determine if ignition occurs.

CAUTION: Suitable precautions shall be taken when making electrical tests of combustible vapors to include at least the use of blow-out panels to preclude explosion of the structure, the location of fire extinguishing equipment nearby and protection of test personnel against possible flame or blast effects.

5.0 CRITERIA TO BE SPECIFIED

a. Waveform

Voltage waveform B of this document shall be applied for this test. The crest voltage shall be sufficient to cause streamering over the exterior of the test object but not sufficient to cause sparkover of the high voltage gap. This will usually require an average electric field of about 500 kV/meter.

b. Number of discharges to be fired.

6.0 TEST PROCEDURE

a. Set up the high voltage generator, discharge circuit and diagnostic equipment.

b. Inspect the equipment and area for safe operation.

c. Fire a series of test discharges to achieve the required test voltage and to check the waveform to assure that sparkover to the test object will not take place.

d. Open the shutter on the camera and fire the first test discharge. For tests of fuel system components the same diagnostic criteria as are specified for the vapor ignition test of Method 302 are applicable.

e. Inspect for streamering data."

"7.0 DATA TO BE COLLECTED

Data shall include:

a. Environmental data such as temperature and humidity.
b. Description and photographs of the test setup.
c. Date, personnel performing the test and location of the test.
d. Test voltage waveforms and magnitudes.
e. Photographs of corona and streamering."

Typical Test Circuits

Voltage waveform B, shown in Figure 1, is the full-wave lightning impulse voltage waveform defined in USA Standard 68.1 (Reference 2) and International Electrotechnical Commission Standard 60 (Reference 3). In this case, a full wave is applied at a voltage level lower than that required to flash over the gap. The voltage is normally generated by a marx-type high voltage generator operating through a series resistance into a capacitive load.

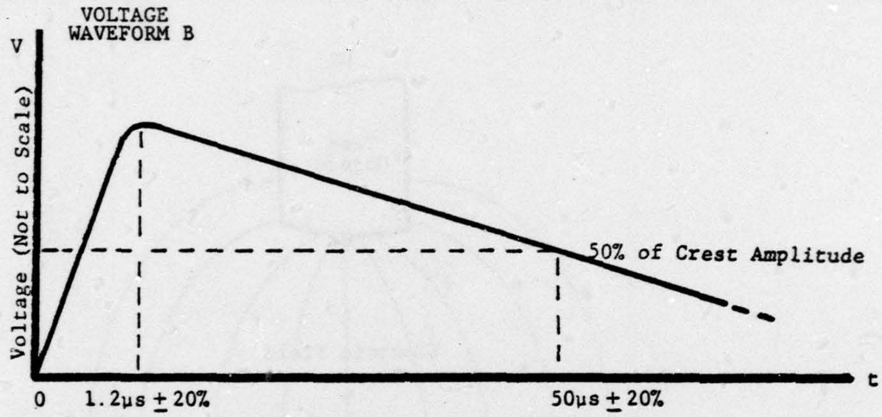
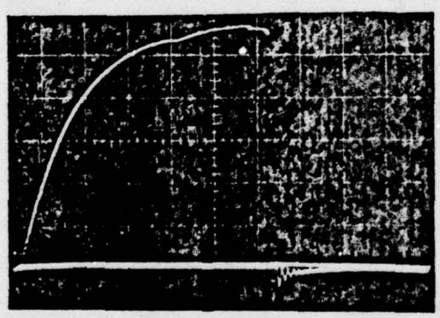


Figure 1 - Voltage Waveform for Streamer Tests (Waveform B).

Whereas waveform B is specified for this test in Reference 1, the actual waveform is not critical. Voltages with longer rise times and correspondingly longer decay times can as well be utilized, with very similar results. Such a waveform is shown in Figure 2.



313 kV/cm 10 μs/cm

Figure 2 - Long-fronted Voltage Waveform for Corona and Streamering Tests.

If the long-fronted waveform of Figure 2 is utilized, corona and streamers will begin to form during the rise and its peak (crest) voltage will have to be set about 30% lower than that of the short-fronted waveform of Figure 1 to prevent complete flashover of the gap. Even so, the probability of gap flashover is increased. It may therefore be advisable to place a parallel gap in the circuit to provide a path for such flashovers, located out of range of the cameras utilized to photograph streamers.

Test Setup

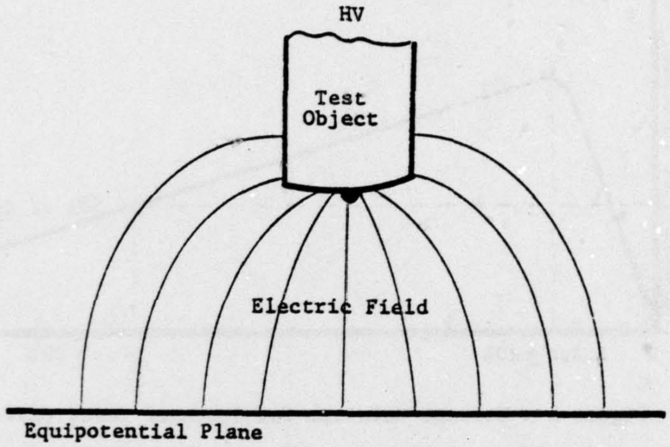
There are two basic methods of applying an electric field to the test object. In the first method the test object is connected to the high voltage output of the test circuit and suspended in proximity to a ground plane or a grounded electrode. In the second method, the test object is at ground potential and a high voltage electrode is used to apply the electric field. In either case the average intensity, E_{ave}, of the electric field about the test object is given by:

$$E_{ave} = \frac{V}{g} \tag{1}$$

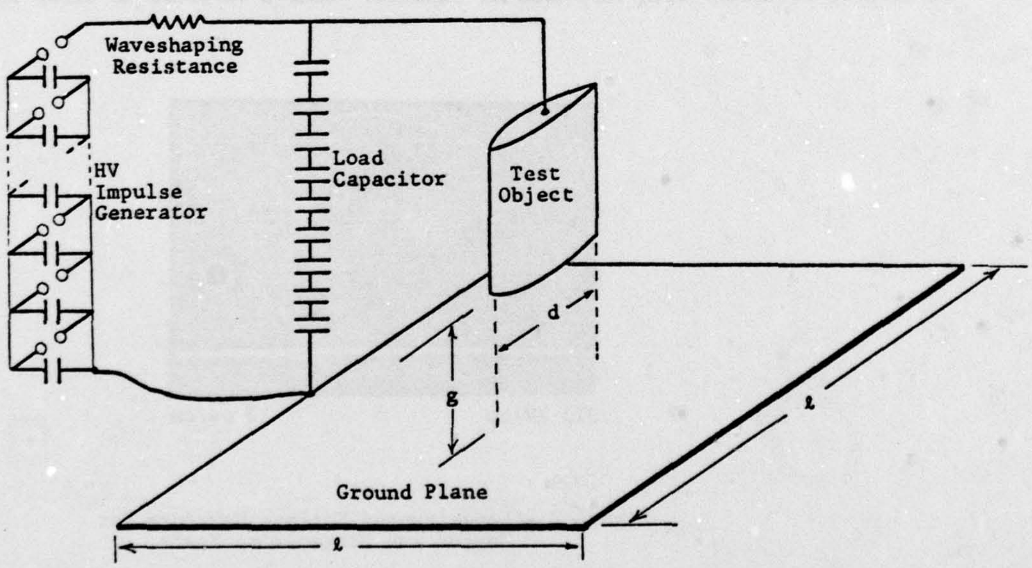
where V is the voltage to which the test object is raised and g is the distance between the test object and the ground plane. Either method will provide adequate streamers. Considerations pertinent to each method are as follows:

Test Object at HV Potential

In this case, the test object must be suspended above (or otherwise in proximity to) a grounded electrode representing an equipotential plane as shown in Figure 3.



a. Situation being represented



b. Test Setup

Figure 3 - Streamering Test with Test Object at High Voltage.

The ground plane must be large enough so that field concentrations about its edges do not influence the field at the test object. Its size, therefore, is dependent on the size of the test object and the distance, g, between itself and the test object. The following relationships have been found to provide a satisfactory test:

$$\ell \geq 5d \tag{2}$$

$$g \geq 2d \tag{3}$$

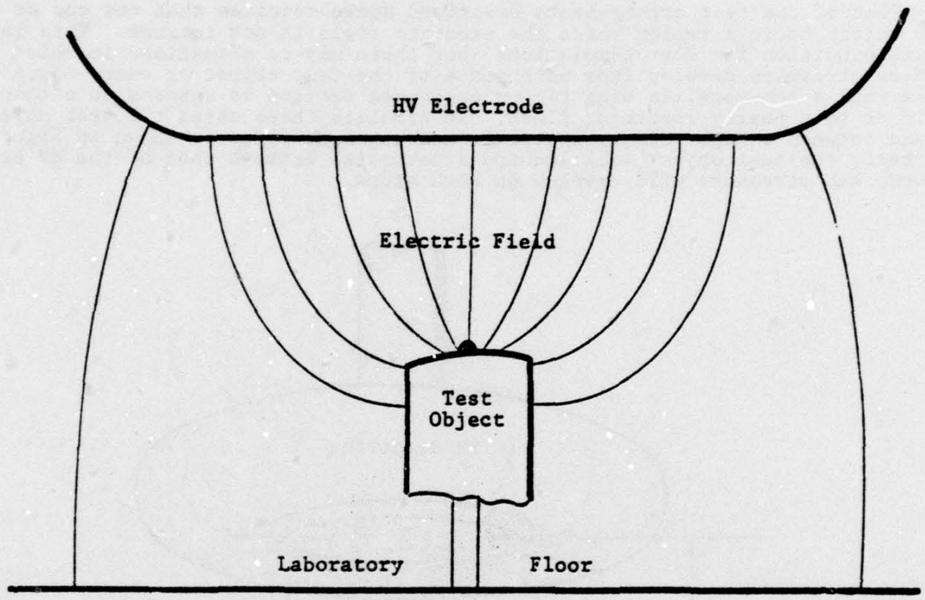
which implies that:

$$g \gtrsim 0.4\ell \tag{4}$$

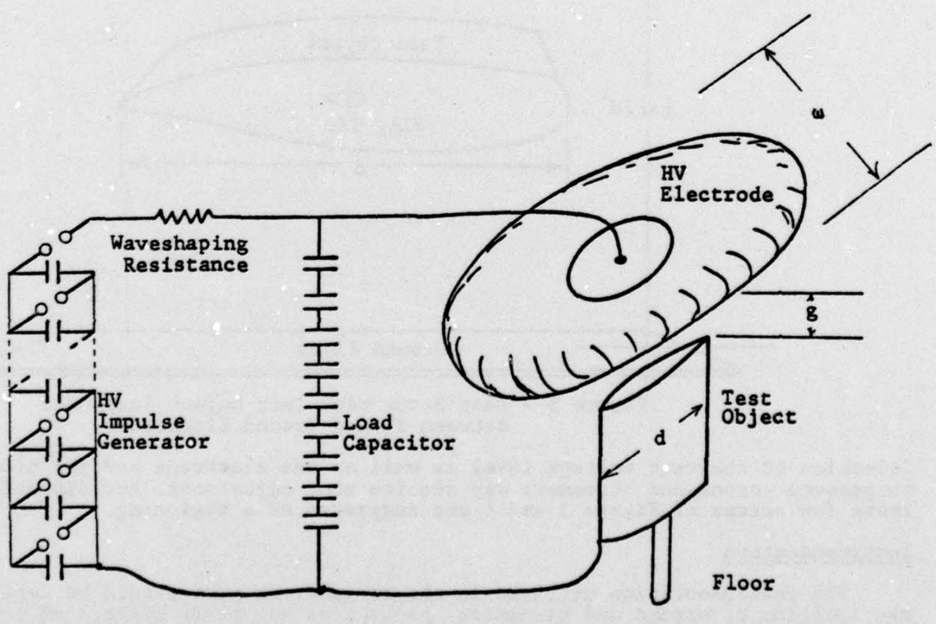
For example, for a wing tip fuel tank of chord d = 1 meter, the gap, g, should be at least 2 meters, requiring a crest voltage of about 1,200 kV to produce corona and streamering.

Test Object at Ground Potential

In this setup, the test object is usually set upon a pedestal and high voltage is applied to a large electrode suspended above the test object. This produces an electric field about the test object as shown in Figure 4.



a. Situation being represented.



b. Test Setup

Figure 4 - Streamering Test with Test Object at Ground Potential.

For the setup of Figure 4, the HV electrode should have sufficiently large radii of curvature in all dimensions to prevent corona and streamers from occurring at its surfaces. The air gap, g, should normally bear the same relation to test object size as discribed previously for the setup with the test object at HV potential. A large automotive inner-tube may suffice for the electrode, provided it has sufficient electrical conductivity to distribute charge over its surface. Otherwise, a smooth metal object with rounded edges may be utilized.

There are advantages and disadvantages of each of the aforementioned setups. If instrumentation such as induced voltage measurement cables or remotely controlled cameras are to be installed within the test object, it may be best to have the test object grounded as in Figure 4. On the other hand, this requires that a relatively large HV electrode be provided. The arrangement of Figure 3 utilizes the test object itself as the high voltage electrode and requires only that a flat conductive sheet or wire screen be laid upon the floor to serve as the ground electrode.

Other Setups

Each of the test arrangements described above requires that one end or side of the test object be in a region where the electric field is not intense. This is the appropriate condition for most simulations, but there may be situations in which it is desirable to have streamers develop from both sides of the test object at once. Such a case might arise when a non-metallic wing tip or empennage section is exposed to a thunderstorm crossfield or to a nearby lightning flash. To simulate these cases the test object may be positioned between a high voltage electrode and a ground plane as shown in Figure 5. During the test, the test object will acquire a potential between that of the HV electrode and ground, and streamers will develop on both sides.

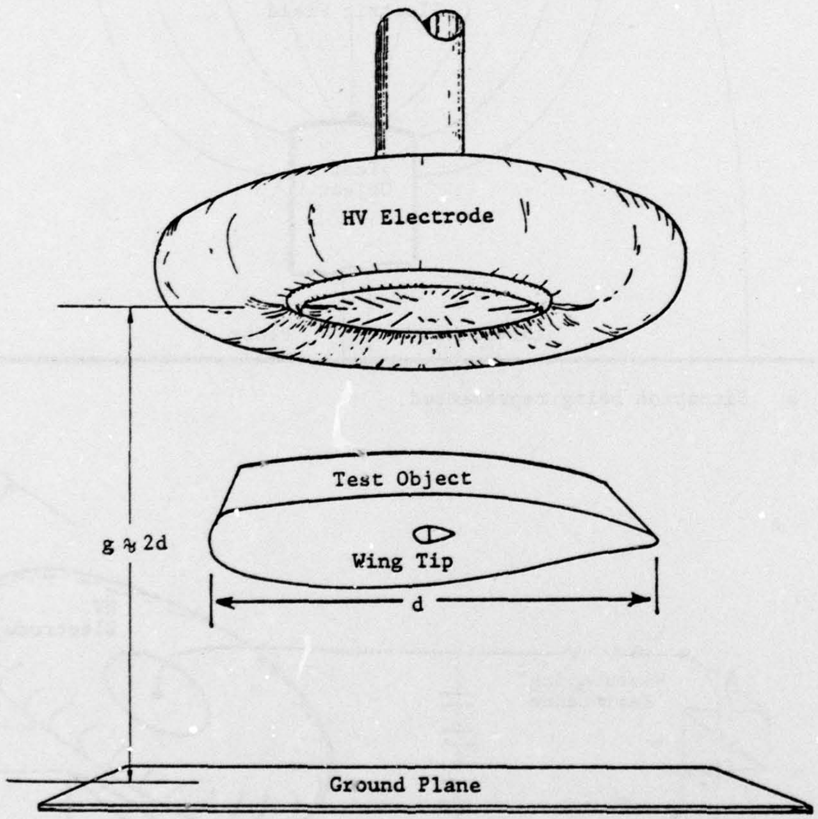


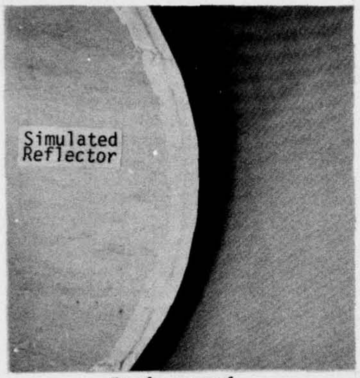
Figure 5 - Test Setup with Test Object Suspended between HV and Ground Electrodes.

Selection of the test voltage level as well as the electrode and gap dimensions necessary to produce corona and streamers may require some adjustment, but dimensions similar to those for setups of Figure 3 and 4 are suggested as a beginning.

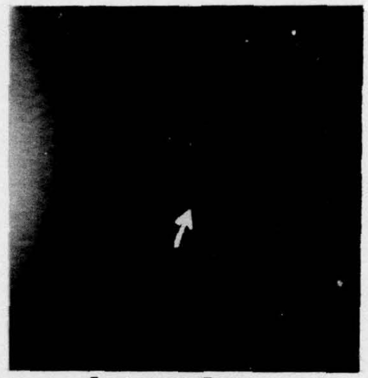
Instrumentation

The instrumentation utilized in the streamering test should be capable of detecting the location of corona and streamers, as well as any other effects of consequence, such as induced voltage transients. As stated in the new standard, still photography with film of 3000 speed sensitivity or more is usually adequate to detect the presence of corona and streamers. The camera(s) should be focused upon suspected corona initiation points on

the external surface and/or within the test object, opened just prior to the HV discharge, and closed immediately afterwards to assure that background light does not overcome the light emanated by streamers. In some cases the light produced by the streamers may be very faint. Because of this it is usually necessary to perform the test in a darkened room and it may not be possible to show the test object background on the same film. A subsequent background photograph under lighted conditions should be taken to assist in identifying the location of streamers visible in the darkened photographs. Examples of streamer and background photographs are shown in Figure 6.



Background



Streamer Test

Figure 6 - Photograph of a Streamer from the Edge of an Antenna Reflector.

The object being viewed is a simulated radar antenna reflector within a fiberglass radome. The antenna, which was made of wood covered with aluminum foil, is visible in the left photograph. A streamer extending from the edge of the reflector to the radome wall is visible in the right photograph. The pictures were taken with Polaroid type 107 3000 speed film with the lens set at F4.7. In this test the antenna was grounded and the electric field was applied from a HV electrode outside of the radome. The pictures were taken by a camera located at the base of the radome.

Some common locations where corona and aperture streamers may be of concern, together with the effects to be on the alert for, are as follows:

- Windshields (puncture, and induced surge voltages)
- Fuel Vent and Dump Outlets and Refuelling Probes (ignition of fuel vapors)
- Fiberglass Radomes, and Wing and Empennage Tips (puncture of fiberglass skins, and voltage surges in enclosed wiring)
- Canopies (streamers from the pilots' head, producing electrical shock)

The streamering test may be utilized to evaluate each of these and other situations in which corona and/or streamers may be a hazard to flight safety or mission reliability.

References

1. "Lightning Qualification Test Techniques for Aerospace Vehicles and Hardware", proposed military standard, SAE Committee AE4L, 20 June 1978.

2. "Standard Techniques for Dielectric Tests", USA Standard C68.1/IEEE Standard No. 4, 1978.

3. "High Voltage Test Techniques," International Electrotechnical Commission Standard 60-2 (1973).

QUESTIONS and ANSWERS

From T.L. Boulay

Q – *What physical situations are represented in the streamering tests ?*

A – This test represents situations where the electric field is intense enough to cause corona and streamering (at areas of concern such as fuel vent outlets or antennas) but not intense enough to result in leader attachment.

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LABORATORY TESTS FOR UNDESIRED CONDUCTED CURRENTS AND SURGE VOLTAGES CAUSED BY LIGHTNING (QUALIFICATION TEST)

by

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Closed Conference
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For Lightning and Atmospheric
Electricity Hazards
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Chatillon, France

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MCDONNELL DOUGLAS CORPORATION

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SUMMARY

Lightning strikes to electrical equipment mounted on the external surface of an aircraft may produce undesired currents and surge voltages on internal wiring, either by direct contact of the lightning arc to the electrical circuit or by electromagnetic coupling due to the intensive fields produced by the lightning strike. These undesired transients may not only threaten the operation of avionics equipment connected directly to the external components, but may even couple electromagnetically into other unrelated circuits and equipment within the airframe.

Laboratory Qualification Tests of aircraft electrical equipment are conducted using high current generators to produce current pulses characteristic of natual lightning. Lightning currents are discharged through the test article and voltages appearing in the associated wiring are measured. This paper describes the equipment, current waveforms and techniques required for these tests.

1. INTRODUCTION

1.1 INTRODUCTION. All aircraft are equipped with numerous electrical devices mounted on the external surfaces of the vehicle. Modern military aircraft are typically equipped with a wide array of lights (navigation, formation, running, in-flight refueling), probes (pitot-static, angle-of-attack, total temperature, stall warning) antennas (communication, navigation, fire-control), and various other electronic devices for defensive and offensive military purposes. Each such device is typically connected by electrical wiring to associated avionics equipment located inside the aircraft. The electrical wiring to and from these components is often bundled together with other wiring into cable runs which may be routed throughout the aircraft.

The most serious incidents on aircraft related to lightning strike damage involve fuel and electrical systems. The greatest threat to the electrical systems is from lightning attachments to externally-mounted electrical components or equipment. These external components provide opportunities for lightning currents and voltages to gain entrance to the interior regions of the aircraft where serious electrical damage may be produced. If proper design measures are not taken, lightning arcs may attach to the external equipment and find their way directly into the electrical wiring and associated avionics. For example, a lightning attachment to a wing-tip navigation light might shatter the protective glass covering or burn through the metallic housing and contact the filament or electrical power leads. The current can then melt or explosively vaporize the wires, and the associated voltage surge may cause widespread breakdown of insulation and damage to electronic equipment.

Design measures can be taken to prevent direct lightning entry into electrical equipment. Lightning arrestors, surge protectors and external diverters or shields may be employed if the inherent properties of the surface equipment do not provide sufficient protection.

In some instances, undesired currents and surge voltages may be produced even though the lightning arc does not directly contact the system wiring. The intense electromagnetic fields produced by the lightning flash may induce transient voltages and currents into the electrical circuits. In such cases, no observable physical damage may be produced, but sensitive electronics may be damaged or temporarily disabled. In either case (direct attachment or induced), transients may also be coupled or transferred to nearby wiring so that other unrelated electrical equipment may also be affected.

The purpose of the laboratory test described in this paper is to qualify externally-mounted electrical equipment by evaluating lightning protective design measures (both inherent and added) and ensuring that hazardous voltages are not produced in the aircraft.

2. LABORATORY TESTING

2.1 GENERAL DESCRIPTION. The objective of this test is to determine the magnitude of voltages and currents produced on aircraft wiring when lightning attaches to externally-mounted electrical hardware. The object to be tested (a production-line hardware component or an accurate prototype) is mounted on a shielded test chamber, so that access to its electrical connectors can be obtained in an area relatively free from extraneous electromagnetic fields produced by the discharge of the high current lightning simulator. The test object is mounted on the test chamber exactly as it is mounted on the aircraft, since normal bonding impedances can contribute to the voltages induced in the electrical circuit. Inside the shielded enclosure, connectors and wiring representative of the flight installation may be used to connect the test object to either the actual equipment or to a simulated load impedance. Usually a dummy load is used since the test is designed primarily to measure the levels of voltage and current appearing on the wires. The measuring instruments may also be located in the same shielded enclosure or in a separate shielded room connected by a suitably shielded instrument cable.

A laboratory generator capable of producing the prescribed lightning waveform is used to inject simulated lightning currents into the test object at the various points where lightning can attach. The test object is grounded via the shielded enclosure so that current flows through the test object in a manner representative of the aircraft installation. The conducted and induced voltages produced in the related electrical circuit are measured at the terminals of the circuits with suitable measuring equipment.

The significant elements of this test which shall be discussed in greater detail include:

a. transient voltage coupling mechanisms
b. the test object and its installation
c. the high current waveforms and their application
d. measurement equipment and techniques

2.2 TRANSIENT VOLTAGE COUPLING MECHANISMS. Voltages or currents appearing in aircraft electrical circuits, as a result of a lightning strike to the aircraft, have all been classified as Indirect Effects. These effects can be further divided into two general categories, depending upon how the voltage was caused. These are:

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a. arc attachment

b. induced coupling.

An arc attachment voltage or current would probably become obvious by the associated physical damage. For example, if the wing tip light lens were punctured by a lightning current and the lightning arc attached directly to the electrical wiring, then physical damage to the lens would be evident and large currents and voltages would appear in the aircraft wiring to the light.

Induced Coupling effects are not so obvious as the arc attachment effects, and to help understand the scope of the problems, the wiring in the aircraft can be classified into two general categories:

a. Airframe return circuits - if the airframe is utilized as a common ground return path, it becomes a part of the signal carrying circuit loop. Induced voltages then result from:

(1) resistive voltage drops resulting from the lighting current traveling through the metal or composite airframe ($e = iR$), and
(2) electromagnetic flux linkage to the signal carrying circuit loop ($e = d\phi/dt$).

b. Two wire circuits - the signal may be grounded at the supply bonding point, but not at the remote component location. Induced voltages or potential differences then result from:

(1) common mode potential differences because the airframe is carrying the lightning current and the signal leads are not,
(2) common mode magnetic flux linkage to the wire/airframe loop (open circuit voltage), and,
(3) differential voltages produced by either magnetic or electric field flux linkage between the two wires of the pair.

Magnetic coupling results because the aircraft skin does not provide perfect electromagnetic shielding for the aircraft wiring. As a result, the lightning current conducted by the aircraft sets up a nonuniform magnetic field inside the aircraft which couples to the aircraft wiring. Magnetic coupling includes both aperture coupling and diffusion coupling. An abvious example of aperture coupling would be coupling to the electrical circuits in the cockpit of an aircraft. Aperture coupling could also occur at joints around the edges of components, access doors or similar nonuniform mechanical points. An example of an aperture coupled open circuit voltage and the driving current is shown in Figure 1. Note that the maximum voltage occurs when di/dt is greatest, i.e., at the start of the current, and the induced voltage is zero when the current reaches its peak (100 kA) because di/dt = 0 at that point.

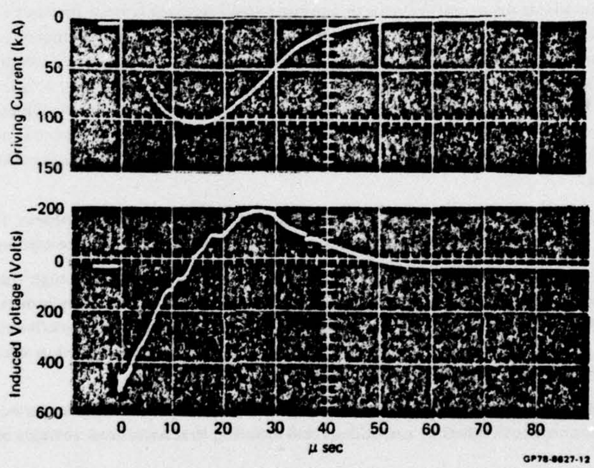

Figure 1 Example of Aperture Coupled Voltage

Diffusion coupling may become more important as more composite materials are employed. Because of the initial high rate-of-rise of current in the lightning strike, the lightning current and associated fields are initially confined to the outer surface of the aircraft skin (where there are no apertures), because of the skin effect; however, as the current continues to flow, the low frequency components diffuse through the skin, and the magnetic field inside the aircraft increases accordingly. As a result, the induced voltage from diffusion coupling will be delayed and will be less than that which would have occurred if the conductive skin had not been there. For composite materials, the diffusion time is much shorter than for metal.

Because of the complexity of aircraft wiring and cable routing, and because of the complex mechanical assembly of the aircraft structures, it is extremely difficult to calculate the voltage and currents on wiring resulting from a lightning strike. Therefore, a transient test might be useful for design verification of the maximum voltages which could appear in critical circuitry. It may be desirable to subject the whole aircraft to the simulated lightning currents because some of the electrical wiring is routed over devious paths and is thus susceptible to diverse fields; such full-scale vehicle tests are addressed elsewhere. However, tests on externally mounted components such as antennas, lights and probes can be conducted using subassemblies.

2.3 THE TEST OBJECT AND ITS INSTALLATION. As mentioned earlier, the test object should be an actual production-line item of the type to be installed on the aircraft mold line; that is, an actual light, probe, antenna or other component and its mounting flange or structure. The associated avionics need not be included, so the hardware expense is mainly that of the external component. Although the associated avionics may be included in the test, if they are not, it is important to simulate the effective impedances across the terminals of the components. The circuit impedance will generally affect the magnitude of induced transients.

In order to properly evaluate the extent of voltages and currents produced on internal circuitry, it is necessary to test the component in such a way as to avoid introducing extraneous transients. This objective is most easily met by mounting the component to be tested on a shielded enclosure with the connecting cables and termination impedances closed up inside. Otherwise, the large fields produced by the discharge of the lightning generator will couple directly into the wiring, producing unrealistic transients.

Generally, an aperture in the shielded enclosure can be fitted with an adapter or mounting plate, which is specially machined with apertures and fastener hole patterns representative of the aircraft skin where the component is mounted. Mounting of the component to the adapter plate is then exactly as it is on the aircraft. The adapter plate is then fastened to the shielded enclosure, using highly conductive RF gaskets to ensure a good EMI seal, and good high current transfer across the interface. All connecting wiring and terminating impedance elements are then located inside the shielded enclosure, which is grounded to the return side of the lightning high current generator.

The configuration of the return conductors is important because they can influence current flow patterns in the vicinity of the test object and, therefore, the amount of induced coupling. The return conductors should be distributed around the test article and spaced a few test article widths away, to ensure minimum impact on the test.

2.4 *LIGHTNING HIGH CURRENT WAVEFORMS.* The lightning threat parameters to which specific components on an aircraft might be exposed is determined by an analysis of the interaction of a severe lightning strike with a moving aircraft. A severe model lightning strike current waveform has been defined and widely accepted for use in the industry. It is shown in Figure 2. The waveform comprises four distinct components, designated by letter as Components A, B, C and D.

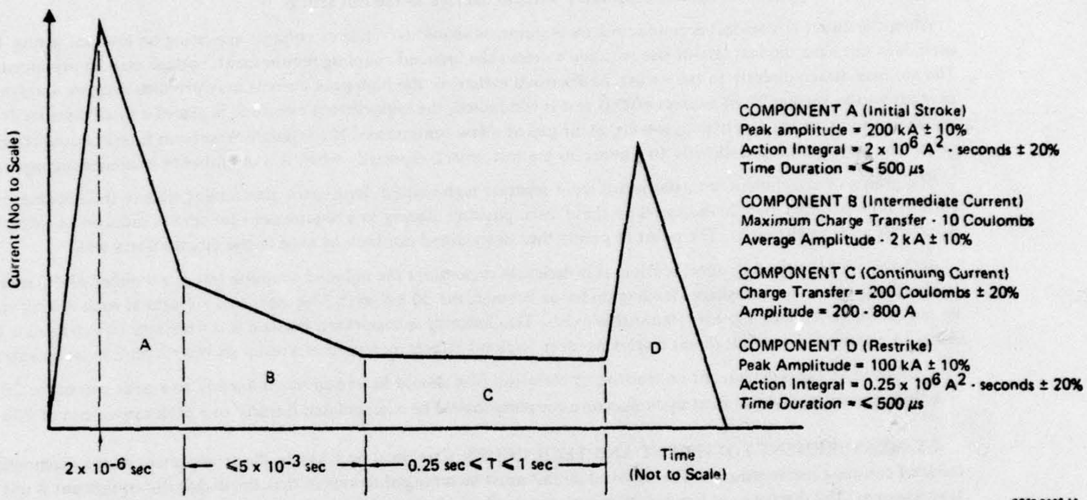

Figure 2 Idealized Current Test Waveform Components for Evaluation of Direct Effects

The waveform components to be used in a given test depend upon the location of the hardware on the aircraft, since the strike may persist for large fractions of a second and the aircraft may move relative to the lightning channel. The table below prescribes the proper applications of the model waveform components for direct structural effects by aircraft zone.

Table 1 - Application of Waveforms for Direct Effects - Structural Tests

Aircraft Zone	Model Current Waveform Components A	B	C	D
1A	x	x		
1B	x	x	x	x
2A		x	x	x
2B		x	x	x
3	x	x	x	

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The aircraft zones each have different lightning attachment or transfer characteristics. These are defined as follows:

Zone 1: Surfaces of the vehicle for which there is a high probability of initial lightning flash attachment.

Zone 2: Surfaces of the vehicle across which there is a high probability of a lightning flash being swept by the airflow from a Zone 1 point of initial flash attachment.

Zone 3: Zone 3 includes all of the vehicle areas other than those covered by Zone 1 or Zone 2.

Zones 1 and 2 are further divided into A and B regions, depending on the probability that the flash will hang on for any protracted period of time. An A-type region is one in which there is low probability that the arc will remain attached and a B-type region is one in which there is a high probability that the arc will remain attached.

It can be seen from Table 1 and from the zone definitions that electrical hardware located in either Zone 1 or Zone 2 will receive either current Component A or Component D, or both. Each of these current pulses in nature may exhibit very fast rates-of-rise, although the rate-of-rise is not specified in the waveform parameters for these components. Rate-of-change of current is an important parameter, since magnetic coupling effects ($e = d\phi/dt$) are directly influenced.

Since the model waveform does not specify rates-of-rise of current, a specialized current waveform designated as Current Waveform E, has been defined for use in tests for induced coupling effects. Current Waveform E is not intended to resemble a particular lightning strike component in terms of energy content, and is not used, therefore, in damage effects testing. It is intended to simulate the early-time initial wavefront of the return stroke current pulse (either initial strike or restrike), where di/dt's (rate-of-change of current) can reach 100 kA/μsec.

As discussed earlier, undesired voltages and current surges can be produced in aircraft wiring by either arc attachment to the wiring or by induced coupling. Therefore, a complete evaluation of Zone 1 or Zone 2 externally-mounted electrical hardware should include tests with the proper waveform components from Table 1, plus a fast rate-of-rise test with Current Waveform E. If the Component A or D waveform actually used in the direct effects test has a fast rate-of-rise conforming to the requirements for Waveform E, the separate fast rate-of-rise test is not required. However, Current Waveform E is often used separately for induced coupling measurements because it is a low-energy waveform and can be applied repeatedly with no damage to the test article.

When the direct effects test is conducted, measurements should be made of voltages appearing on internal wiring. Even if the waveform used does not have the fast rate-of-rise required to meet the induced coupling requirement, voltage may be produced by other mechanisms. The arc may attach directly to the wiring, as discussed earlier, or the high peak current may produce resistive voltage drops which can appear across the wiring. When a direct effects test is conducted, the high current electrode is placed a small distance from the desired arc attachment point on the test object, leaving an air gap of a few centimeters. If a separate Waveform E test is conducted, the output of the generator may be clamped directly to a point on the test object, especially when it is desirable to minimize damage to the test article.

The points of attachment are determined from separate high voltage, long-spark attachment tests to the full-scale components. Numerous current flow patterns may be indicated by these tests, possibly leading to a requirement for several induced coupling tests to determine the worst-cast attachment point. The point or points thus determined can then be used in the direct effects test.

Although not required by specification, it is desirable to conduct the induced coupling test at a number of different peak current levels (maintaining the same waveshape) leading up to, or beyond, the 50 kA level. The measured voltages at each driving current level can then be plotted to verify that a linear relationship exists. This linearity is important, because it is necessary to extrapolate the measured transients to correspond to a full threat level waveform. Indirect effects measured as a result of Waveform E must be extrapolated as follows:

a. Induced voltages dependent on resistive or diffusion flux should be extrapolated linearly to a peak current os 200 kA.
b. Induced voltages dependent upon aperture coupling should be extrapolated linearly to a peak rate-of-rise of 100 kA/μsec.

2.5 MEASUREMENT EQUIPMENT AND TECHNIQUES. Care must be taken in the arrangement of the diagnostic equipment used for induced coupling measurements. The physical circuit must be arranged to ensure that the diagnostic equipment is not influencing the system response. The discharge of laboratory generators produces intense electromagnetic fields over a wide range of frequencies. High frequency RF energy from spark gap switches and rapidly changing electric fields accompany the magnetic fields produced by the high peak current surges. This intense EM field environment will induce currents in any exposed electrical conductors, including exposed signal leads, building power lines, and measuring instruments. These spurious signals can easily mask the desired signal if care is not taken in the design and installation of the instrumentation system.

In addition to the problem of shielding the instrumentation system, the frequency response of the total measurement system must be fast enough to detect the very sharp high frequency transients which often occur at the instant the generator is triggered. For most small test articles, a frequency response of 10 MHz is probably sufficient. However, for some very large test articles which are complete subsystem in themselves (such as an external fuel tank), the natural resonant frequency of the structure and of internal wiring may be excited. Therefore, the frequency response of the system should be fast enough to record any such transients (up to 30 MHz).

In addition to the instrumentation itself, if long shielded cable runs are employed, they may load down the signal also, both in frequency and magnitude. Care must be taken in the design of impedance matching devices and other interface circuitry to ensure that the signal is not distorted or masked.

The instrumentation may be installed in the same shielded enclosure on which the test object is mounted. Such an installation requires adequate volume in the enclosure, as well as good ventilation for cooling and quick access for data retreival. The advantages of this arrangement are that the measurement leads are short, and minimum additional effort is required in shielding of the instruments.

In other instances, the instrumentation may be housed in a separate shielded room. The measured transient signals must then be transmitted to the shield room via signal leads of some type. In this case, precautions must be taken to ensure that spurious signals are not induced in the signal leads. Such measures might include:

a. Double shielded wire for the instrumentation cable.
b. Double shielded duplex cable and differential readout instrumentation to eliminate common mode voltages.
c. Completely floating battery powered instrumentation (elimination of ground loops and power line filtering).
d. Fiber optics to isolate the readout instrumentation from both the electrical circuit under test and the transient generator (elimination of ground loops).
e. Fiber optics and very short instrumentation cables to minimize loading effects on the test circuit.
f. Isolation resistors at the pickoff voltage point to minimize the loading effects on the circuit under test.

Even after all these precautions have been taken, the elimination of noise in the measurement instrumentation system should be verified. This can be accomplished by delivering the test current to the shielded enclosure while monitoring the readout instrumentation with the leads disconnected from the electrical circuit under test. This should be done under two conditions:

a. with the disconnected measurement leads open circuited, and,
b. with the disconnected measurement leads shorted.

2.6 INTERPRETATION OF DATA. As discussed earlier, induced voltages produced by Current Waveform E must be extrapolated to threat levels. Specifications such as MIL-B-5087 allow transient voltages up to 500 volts in magnitude on internal wiring. In modern technology aircraft, much lower voltages could be troublesome in some sensitive avionics. Therefore, the proposed international standard does not attempt to specify what magnitude transients are allowed. That determination is left to the cognizant regulatory agency and the test establishment. Efforts are now underway in the United States to develop a new government/industry standard for the treatment of aircraft transients. The new standard will hopefully address the problem from both aspects, i.e., the susceptibility of equipment and the shielding provided the aircraft installation.

QUESTIONS and ANSWERS

– From G. Orion

Q – Are the lightning tests mentioned by Mr. Clifford becoming mandatory in new U.S. contracts ?

A – The contracting agency determines whether the aircraft they are purchasing must perform in a thunderstorm environment. If the answer is yes, then a lightning specification is imposed on the contractor. In the past, the primary specification for that purpose has been MIL-B-5087B, which does require that all external electrical hardware undergo laboratory tests for lightning damage evaluation. We have conducted such tests on our aircraft since 1970. The MIL-B-5087B test requirement is not significantly different from the test described in this paper. The primary difference is that the new proposed standard specifically allows testing with current waveforms determined by zonal location of the hardware on the aircraft. The new standard also provides for a separate induced coupling test using a fast rate of rise, low energy current waveform (Current Waveform E). Since the new proposed standard has not yet been offically adopted in the U.S., the specific test program described in this paper is not at this time a mandatory requirement. However, when and if it does become an official specification, the impact will be small for this particular class of test because of the close similarity to the MIL-B-5087B requirement.

LABORATORY TESTS FOR UNDESIRED CONDUCTED CURRENTS AND SURGE VOLTAGES CAUSED BY LIGHTNING (QUALIFICATION TEST)

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SUMMARY

The theoretical background for the extrapolation of induced voltages and currents measured during the test is described, and sample waveforms are shown. Guide lines are given for recognising the appropriate coupling mechanism causing the induced voltage in order that the correct scaling may be applied. The choice of simulated impedances is discussed in consequence of the broad frequency spectrum of lightning pulses.

Introduction

In Test 401, paragraph 8.0 (Notes) section (b) it is stated that indirect effects measured as a result of waveform E should be extrapolated linearly to a quoted peak current $\hat{I}$ (=200 kA) or to a peak rate of rise, $\widehat{\frac{di}{dt}}$ (= 100 kA/μs.)

This paper will describe briefly the two principal mechanisms which generate induced voltages and currents, and show how the measured waveform may be analysed for determining the correct scaling factor to apply. The considerations which govern the design of terminating impedances for cables from test specimens are also given.

Resistive/Diffusion flux Induced Voltages

When the construction of the test object is such that the simulated lightning current produces voltages arising from resistive volt drops in the test object or its mounting system, then the voltage so generated will be related to the current amplitude and waveshape. Extrapolation should therefore be up to 200 kA. That is, voltages and currents measured with a test pulse of say 50 kA peak, should be scaled up 4 times to give the equivalent 200 kA value.

The typical waveforms produced by resistive processes (and this includes voltages induced in loops within enclosed conducting structures by diffusion flux coupling - an important phenomenon with carbon fibre structures) are shown in fig 1.

The important characteristics of these voltages are:-

(1) There is no instantaneous jump in voltage at t = 0+; the waveform starts at zero and may then (a) commence to rise at finite slope similar to the current waveshape (especially in very resistive materials like carbon fibre or thin wires) or (b), show a dead time (i.e. zero slope) for a short period before rising (as would be observed in high conductivity materials like aluminium.)

(2) Peak voltage does not occur at t = 0+; but will normally occur at or near peak current, often somewhat early when measured within carbon fibre/metal structures; and in high conductivity materials may occur late owing to the time delay introduced by the diffusion process. (Examples of these waveforms are shown in fig 1.)

Fast flux ('aperture') coupling

When the magnetic flux surrounding conductors carrying the simulated lightning pulse, or some fraction of it, couples with loops (e.g. in unscreened pitot heater wires, aerials etc,) then the induced voltages will be proportional to $\frac{d\Phi}{dt}$ in that loop.

The flux external to a conductor, and flux within apertures having insulating covers is instantaneously proportional to the current, and hence $\frac{d\Phi}{dt} \propto \frac{di}{dt}$

Therefore the voltage waveform will normally bear a strong resemblence to the first derivative, $\left(\frac{di}{dt}\right)$ of the current waveform, and therefore extrapolation will be up to 100 kA/μs. That is, an induced voltage of di/dt type caused by a current pulse whose peak di/dt is say, 40 kA/μs should be multiplied by 2.5 to give the 100 kA/μs value.

The important characteristics of these waveforms are:-

(1) A fast step in the waveform amplitude occurs at t = 0+, at the commencement of the current pulse. The waveform is often accompanied by 5 to 50 MHz components, and the first zero crossing will be at peak current where $\frac{di}{dt} = 0$

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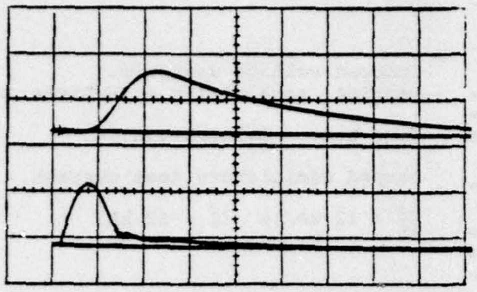
(2) The peak voltage attained will normally occur at the initial transient at $t = 0+$. (If peak voltage occurs at or near peak current the mechanism cannot be aperture coupling). Typical waveforms are shown in figure 2.

Choice of simulated load impedances for indirect effects tests

The simulated load impedances should be chosen to make the effective load presented to the test object wiring as representative as possible over a very broad frequency range. Where possible, actual circuit components should be used, or replicas thereof, so that low, medium and high frequency impedance is duplicated. It is not sufficient to load a 50Ω aerial cable with 50Ω. The radio receiver/transmitter unit will normally only match the cable with 50Ω over a very small frequency range. During a lightning test the circuit might act like a very low impedance to low frequencies if the receiver includes a shunt choke to ground, or be very high impedance if there is only a small series capacitor. Therefore, some knowledge of the real circuitry into which the feeder operates is required so that meaningful measurements might be made.

Acknowledgement

The author gratefully acknowledges the support given by the Procurement Executive of the UK. Ministry of Defence for this work.

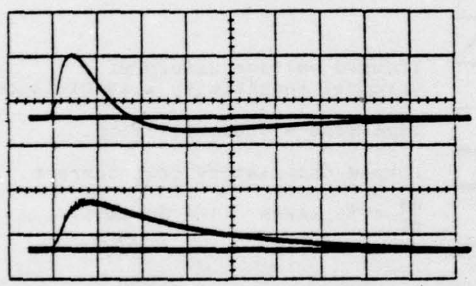


Induced voltage waveform.
Vertical sensitivity = 4 V/Division.

Time base = 10μs/Division.

'Divert' test current waveform.
$\hat{I}$ = 112 kA.

a) 1 inch O.D. Copper Tube (high conductivity material).
Note (i) Voltage waveform 'dead time'.
(ii) Voltage peak occurs much later than current peak.

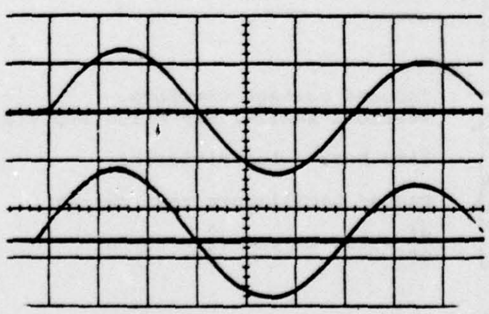


Induced voltage waveform.
Vertical sensitivity = 51 V/Division.

Time base = 1μs/Division.

'Unipolar' test current waveform.
$\hat{I} \simeq$ 8 kA.

b) Composite Structure (Carbon Fibre/Aluminium).
Note (i) Voltage waveform exhibits no discernable dead time but slope of rise is finite (closely follows current waveform rise).
(ii) Voltage peak occurs slightly earlier than current peak.



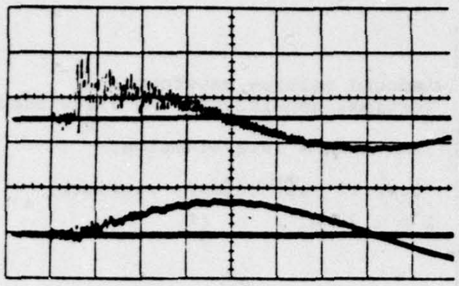
Induced voltage waveform.
Vertical sensitivity = 185 V/Division.

Time base = 0.5μs/Division.

Damped oscillatory test current waveform.
$\hat{I}$ = 37 kA.

c) 0.5 inch O.D. Stainless Steel Tube (lower conductivity than a).)
Note (i) Voltage waveform dead time only just discernable.
(ii) Voltage peak occurs slightly later than current peak.

Fig 1. Examples of Resistive/Diffusion Flux Induced Voltage Waveforms.



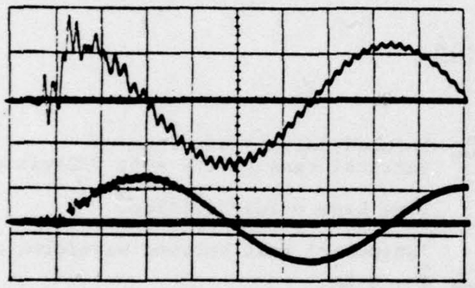
Induced voltage waveform.
Vertical sensitivity = 1V/Division.

Time base = 0.5μs/Division.

Damped oscillatory test current.

$\frac{\hat{di}}{dt} \approx 12$ kA/μs ($\hat{I} \approx 12$ kA)

a) $\frac{dB}{dt}$ measurement - current distribution around aircraft fuselage. Note high Q, 25 MHz transient response.



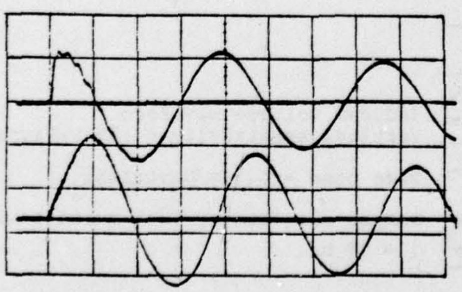
Induced voltage waveform.
Vertical sensitivity = 4V/Division.

Time base = 0.5μs/Division.

Damped oscillatory test current.

$\frac{\hat{di}}{dt} \approx 56$ kA/μs ($\hat{I} \approx 30$ kA.)

b) As a), but different test conditions. Note predominant transient response frequency ≈ 9 MHz (λ/4 mode of load assembly).



Induced voltage waveform.
Vertical sensitivity = 170 V/Division.

Time base = 1μs/Division.

Damped oscillatory test current.

$\frac{\hat{di}}{dt} \approx 100$ kA/μs ($\hat{I} \approx 58$ kA)

c) Voltage on Internal Fuselage wire located behind aperture covered by insulated carbon fibre panel. Note that carbon fibre panel does not carry main structure current and so L.F. voltage is still $\frac{di}{dt}$ type; the H.F. transient, however, is severely attenuated.

Fig 2. Examples of fast flux ('aperture') induced voltages, characterised by fast step at t = 0+, and first zero crossing corresponding to peak current.

QUESTIONS and ANSWERS

1 – From A. Fa'arman

Q – In your circuit showing parallel paths in a composite metal and carbon fibre structure you indicate only equivalent inductances and resistances. Are there any capacitive effects ?

A – Current sharing between metal and carbon fibre in all the circumstances I have studied has not been affected by capacitance. Current sharing is dominated by the time constants associated with inductance and resistances ; the capacitive effects are too small. However some mechanisms of coupling into internal wiring may be partly capacitive, so capacitive effects should not be ignored in induced voltage studies.

2 – From R.H. Evans

Q – I should like to ask a question about extrapolation.

In practice the measured voltage may be a combination of resistive voltage, magnetically induced $d\phi/dt$ voltage, and superimposed HF oscillations. How are these effects separated to extrapolate the measurements to full threat level ?

A – Tests often produce very complex waveforms. However since $d\phi/dt$ voltages peak at the instant of starting of the current pulse, the amplitude here is unaffected by i x R effects and extrapolation is easy. At peak current of the test pulse, $d\phi/dt = 0$, therefore the observed voltage must be resistive, and scaling can be based on the amplitude of the voltage at peak current. HF oscillations can usually be ignored in component tests because they will often be facility effects. However in some cases they may be important, and their relevance should be closely examined.

BIOGRAPHICAL NOTE

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Educated at King Edward VI School Bury St Edmunds and University College London. B. Sc (Eng.) Upper 2nd Class Honours Degree (1958). Chartered Engineer, Member of the Institution of Electrical Engineers. Engineer with GEC Avionics Systems, British Broadcasting Authority, and since 1964 Senior Scientific Officer in the UKAEA Culham Laboratory. Experience at Culham has included work on ion sources and supersonic nozzle sources for clusters ; high voltage, high current switches ; cryogenic systems, and since 1973 section leader of Induced Voltage group of the Lightning Studies Unit.

TESTS ON ACTUAL AIRCRAFT FOR ELECTROMAGNETIC EFFECTS (ENGINEERING TESTS),

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SUMMARY

The paper discusses the requirements for a ground test on an aircraft such that an adequate simulation of the in-flight lightning strike is achieved. However, lack of quantitative data prevents electric fields from being fully simulated. Both time-domain and frequency domain methods are discussed, and it is shown how driving point, earthing point, return conductor and capacitor bank design may be optimised for good simulation. In section II, the Culham quasi-coaxial system is described as applied to the Hawker Hunter test, and more recently to a composite forward fuselage test at General Dynamics.

Section I Problems and Solutions in Simulation Techniques

The objective of these tests is to determine the level of induced voltage and current in the electrical systems within a complete aircraft - or major sections thereof- primarily to identify system incompatabilities. Whilst an aircraft's sub-systems may be tested in the laboratory to the full lightning severity, it is not always easy to establish how the sub-system will react when in the airframe environment, and when in close proximity to the other systems. Also in many cases the airframe is so complex that calculations of the severity level to be expected in any particular system are of limited accuracy and cannot easily anticipate coupling levels into other systems.

Hence, a full aircraft test will provide important supplementary information for the systems integration programme on a new aircraft, or for a new system installation in an existing airframe. Whole aircraft tests are being carried out also as research projects to establish coupling mechanisms and related phenomena. The same general principles apply to the design of such rigs although the design criteria are somewhat different. The difference comes from the greater need during an applied research investigation to simulate accurately the in-flight environment, so that the research effort can concentrate on phenomena related to the real in-flight conditions and not be diverted by unwanted facility effects. Thus the purpose of any particular test will dictate the importance attached to the accuracy of simulation achieved in the test rig design.

Accuracy of simulation includes the following aspects:-

(I) Current level, that is, whether full threat or scaled down amplitude and rates of change are used.

(2) Accuracy of magnetic fields and current distribution around the airframe produced by the lightning current.

(3) Accuracy of electric fields - (not fully described and full quantitative data is not available yet).

(4) Simulation of h.f. structure resonances, which give longitudinal variation of electric and magnetic fields, appropriate to the in-flight condition.

(5) Waveform of test pulse.

Additionally frequency domain as distinct from the more familiar pulse technique may be employed for coupling investigations, particularly at higher frequencies (>0.1 MHz).

Considerations applicable to the points (1) to (5) are as follows:-

1. Current Level

A high level test (200kA peak pulse amplitude and 100kA/µs rate of rise) is desirable since induced voltages so obtained do not need scaling up. Scaling up is dependent on the system remaining linear which cannot necessarily be assumed. If a lower level test is required, the smaller the scaling factor, the better the reliability of the scaled up induced voltage. A high level test poses a hazard to an airworthy aircraft from explosion of fuel tanks, fire in hydraulic systems or damage to electrical equipment. Hence elaborate precautions (including inerting of fuel tanks) may have to be taken on an airworthy aircraft if tested at full level. A high level test will always require a low system inductance owing to the prohibitively large voltages required to give high currents and high rates of rise through high inductance systems. Peak current and peak rate of change of current are given by equations (1) and (2).

$\hat{I} = V K_1 \sqrt{\frac{C}{L}}$ amperes (1) $\qquad$ $\widehat{di}/dt = \frac{V}{L}$ amperes/second (2)

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Where V = capacitor bank erected voltage.
L = total circuit inductance.
C = effective bank capacitance.
K_1 = current factor, which is a function of $\frac{4L}{R^2C}$
where R is the effective series resistance.
When $R \rightarrow 0$ $K_1 \rightarrow 1$. For critically damped pulse
$K_1 = 0.368$.

2. Accuracy of magnetic fields and currents

Longitudinal current flow along a fuselage or a wing produces a magnetic field around the fuselage (or wing) at right angles to the current flow, so that in cross section a circular fuselage would have concentric rings of flux around it in the plane of the cross section. In a non-circular cross section the surface current will be strongly dependent on local surface curvature. The accuracy of the magnetic fields and current distribution around an airframe during a ground test depends upon the position of the return conductor(s) - that is on the position of the conductors which lead the airframe current back to the capacitor bank (or other energy source) to complete the circuit, since these conductors influence the magnetic field. In 'free space' (i.e. in the in-flight condition) the magnetic field around a typical modern military fighter fuselage might be as in figure 1. Large current densities occur where the field lines are close to the airframe (e.g. the strake edges) and lower current densities occur elsewhere. An effective simulation of this is provided by any of the three alternative systems on the right of figure 2. The three systems, I will call full coaxial, multiwire coaxial, and quasi-coaxial. The field around the fuselage of figure 1. when using the 3 broad sheet quasi-coaxial system is shown in figure 3. This closely resembles the field pattern shown in figure 1. close to the fuselage. Closer return conductor spacing should be avoided because of the effects on field distortion in apertures like the cockpit.

The effect of a single return conductor close to the fuselage is illustrated on the left of figure 2, where current is caused to crowd on the surface of the fuselage adjacent to the return. All the systems on the right of figure 2, have a low system inductance (0.15 - 0.2μH/m) so facilitating high level testing. However a low level test may achieve adequate field and current distribution by using say 2 fairly wide spaced returns as in the L.S.T test to be described. The inductance of two widely spaced returns is high so restricting the technique to low or moderate severity testing.

3. Accuracy of electric fields

The accuracy of electric fields produced in ground tests compared to those produced when a lightning arc channel contacts and drives current through the airframe cannot be fully analysed since little reliable quantitative data is available of the in-flight electric fields. To a certain extent, however, the correct simulation of magnetic fields will ensure that electric fields associated with the changing magnetic fields are reproduced.

Electric fields are produced by a changing current between the return conductors and the airframe owing to the inductance of the system. The voltage so produced will vary in a linear manner from zero, at the connection point of the returns to the fuselage, to maximum at the drive end. These electric fields, for a given di/dt, are independant of the return conductor spacing (in fig. 2 right) since both L and the electric stress $\hat{E}$ vary as $\log_e \frac{r_2}{r_1}$. The magnetic field lines of figures 1 and 3 are also equipotentials for the electric field calculation.

4. H.F structure resonances

H.F. structure resonances occur on an aircraft when subjected, for example, to N E M P. The resonances depend on the structure dimensions(length, wing span etc.) and the predominant modes will correspond to $\lambda/2$ frequencies associated with these dimensions plus harmonics. Similar effects will occur on an aircraft when subjected to the electric fields from a nearby thundercloud, nearby strikes and from lightning current flow through the airframe. In all these cases the same structure resonance frequencies will be observed since they will be excited by the field and current transients.

An ideal simulation system will be one in which the frequencies, amplitudes and Q of resonances excited by the transient current pulse will correspond to those observed in a 'free space' condition. Fig 4 illustrates the lightning attachment to a fuselage and the standing wave pattern of the $\lambda/2$ resonance, f_1. Fig 5 illustrates how in a test of one wing, or half the fuselage, this frequency can be duplicated by a $\lambda/4$ shorted quarter wave line set up. Thus longitudinal variations in currents and voltages on the fuselage will occur, as well as the circumferential variations associated with the fuselage (or wing) cross section shape. Return conductors, positioned as in figure 3 for reasons stated in paragraph (2) above will be well placed to give good simulation of the h.f. current distribution also. However care must be taken to avoid changing the electrical length of the aircraft or the resultant frequency may be much lower and therefore not representative of an in-flight strike condition. For this reason the capacitor bank,ground plane and ground power-cart connection (if used) should be so designed that least effect on the natural frequencies occurs. Referring to figure 5, ideally the capacitor bank should be close to point B and small in physical size (with respect to aircraft). System grounding should be at A at which point the power-cart,and diagnostic connections may be made with little or no effect on the system.

5. Waveform of test pulse

This topic has been relegated to last position since the waveform of the test pulse is not very important for testing, provided the current amplitude and rate of rise are measured; and induced voltages scaled up appropriately.

Both sine wave and unipolar (unidirectional) pulses are suitable, since both contain a very wide spectrum of frequencies owing to the transient start at t_o+. (A suddenly starting damped sinusoid has a very broad spectrum and must not be confused with a C.W. sinewave which comprises one line only in the spectrum). Fig 6 illustrates the spectra obtained with a unipolar pulse (upper) or damped sinusoid (lower). For a capacitor bank and load system of fixed V,L and C (as defined in paragraph (1)) the absolute magnitude of the two spectra are identical except around the centre frequency f_o given by $f_o = \frac{1}{2\pi\sqrt{LC}}$

Fig 6 shows the spectra of both the current pulse and of the differentiated current pulse. These two spectra are important when referring to resistive induced voltages (I spectrum), and $\frac{d\Phi}{dt}$ induced voltages $\left(\frac{di}{dt}\text{ spectrum}\right)$.

An alternative approach to data collection, in frequency domain form, is shown by the swept C.W. method, which facilitates the measurement of transfer functions of skin current to internal wiring over a broad frequency range. This is a low level test, but similar considerations apply to the design of rigs for this test technique as described in sections (2), (3) and (4), owing to the need for correct simulation of fields and current distribution. Considerable computer processing of the data is required to convert it to time domain form.

Features of the various test methods

AFFDL/FES L.S.T. (and LTA)	Culham Quasi Coaxial System	Boeing swept C.W
•Time domain (pulse) •Low to moderate severity •Uses standard capacitor bank and simple return conductor configuration, high inductance. •Spurious frequencies introduced by facility layout, ground capacitance and power-cart connections. •Little or no computing necessary (but can be used with advantage). •Short setting-up time.	•Time domain (pulse). •Up to full severity. •Uses small bank and return conductor system designed and built specially for test object. •Allows good simulation of h.f. resonances, particularly for one wing, or half fuse-lage tests. Good control of stray capacitances. •Little or no computing necessary - but field profile calculations are useful for rig design. •Return conductors have to be constructed around aircraft.	•Frequency domain. •Low level techniques. •Uses wide range R.F. oscillator and amplifier and simple return system. •Allows good simulation of h.f. resonances, particularly for one wing or half fuselage tests. •Computing essential to convert measured responses to time domain form. •Return conductor has to be tailored to the test object.

Section II Description of Culham Quasi-Coaxial Test System (C.Q.C.)

1. Original System at Culham

The test system comprises the load assembly (Hawker Hunter fuselage plus return conductors), interconnecting bus bars, and the capacitor bank itself which was a Marx generator, capable of being operated with any number of stages up to 5..

Principal characteristics and parameters of 5 and 1-stage operation are summarised:-

	5 stage 91kV/stage, 455 kV erected	1 stage 100 kV charge
L	4.5μH	2.85μH
C	0.075μF (5 x 0.375 μF in series)	0.375μF
f_o	270 kHz	150 kHz
$\hat{I}$	56 kA	34 kA
$\frac{\hat{di}}{dt}$	100 kA/μs	35 kA/μs.

Fewer stages give a lower inductance, but a much larger capacitance, hence the frequency is lower with 1 stage than with 5. The low overall inductance arises from the low inductance of the load assembly (≈ 1μH) and the low inductance achieved in the bus bars and Marx through the use of broad, close spaced conductors. A cross section and diagramatic side view of the load assembly are shown in figures 7 and 8, and a photograph of the system is shown in figure 9. Diagnostics were by balanced twin screened cables which entered the aircraft at the rear screening cover and then ran in a tube along the earthing bus-bar back to "experimental earth" (system grounding point)

and on to the screened room,within copper tubing the whole way. Important achievements and features of this rig were:-

(a) This was the first test at full threat $\frac{\hat{di}}{dt}$ (100 kA/μs) on a very large component.

(b) First demonstration of high di/dt with 1 stage bank (35 kA/μs). 64 kA/μs was obtained subsequently on single stage operation on the same rig.

(c) Demonstration of free space inductive current distribution around the fuselage with quasi-coaxial system; good agreement was obtained with predicted results.

(d) Demonstration of linear law for scaling aperture flux induced voltages with peak di/dt over the range 10 to 100 kA/μs.

(e) Principal h.f. resonance modes on the load assembly were investigated.

(f) H.f. resonances produced in single stage bank operation were simpler, and more consistent than in multistage operation.

(g) Several high frequencies were observed which were not apparently occurring in the load assembly. This pointed the need for a simpler bank and switch system having less strays (capacitance etc.) and a more compact construction.

II Development of C.Q.C. for General Dynamics Test

The General Dynamics composite forward fuselage test was designed jointly between General Dynamics and Culham personnel. It was decided to use the C.Q.C. system, and to refine it further to achieve the best simulation possible for induced voltage coupling studies on the composite airframe. All of the points mentioned in the theoretical introduction in section I (paragraphs (1) to (5)) were considered at the design stage.

The system as built is shown in fig 10. A single stage bank is used, comprising one 4μF 50 kV capacitor, an arc switch; a 1.35Ω resistor, and the load assembly. Total circuit inductance ≃ 2.25μH (of which ≃1μH was in the load assembly). A charging voltage of 45 kV produced ≃ 20 kA/μs, with 40 kA peak using a lightly damped sinusoidal pulse, or 20 kA with a unipolar pulse. Therefore scaling factors of 5 were required to scale up to 'full threat' values (100 kA/μs, and 200 kA). Some tests were performed at over 100 kA with a bigger bank, and demonstrated that non-linear processes (arcing) on the fuselage occured.

The electrical length of the fuselage was increased aft to bring it to approximately one half the full fuselage length, and the return conductors were fixed there in a manner similar to figure 5.

A compact capacitor/switch/resistor assembly was supported close to the forward end but well clear of the ground. The trigger and charge leads - which would otherwise have provided a second path to ground - had resistive decoupling to damp out h.f. currents along them.

System grounding was at the aft end, where the diagnostic connections were made (through a 10 cm diameter aluminium tube) and 400 Hz power could also be fed in. The system was free of bank, trigger, or other spurious h.f. resonances and only two frequencies, both attributable to the fuselage, were noted. These were at 14 MHz and 46 MHz and were identified as resonances which would occur on a full fuselage in the in-flight condition. Owing to the earthing (grounding) concept utilising the one aft connection described above, no spurious signal was introduced into the aircraft by the power leads, nor into the diagnostics, and a very high signal-to-noise ratio was achieved. Neither were any aircraft frequencies affected by the power connection. The Q of the fuselage resonances could be easily adjusted to a lower value by damping resistors, as shown in figure 8.

These several features greatly improve the usefulness of the test technique. Many or all of the features employed in this design could be used for subsequent whole aircraft tests in order that an improved simulation is achieved.

Acknowledgement

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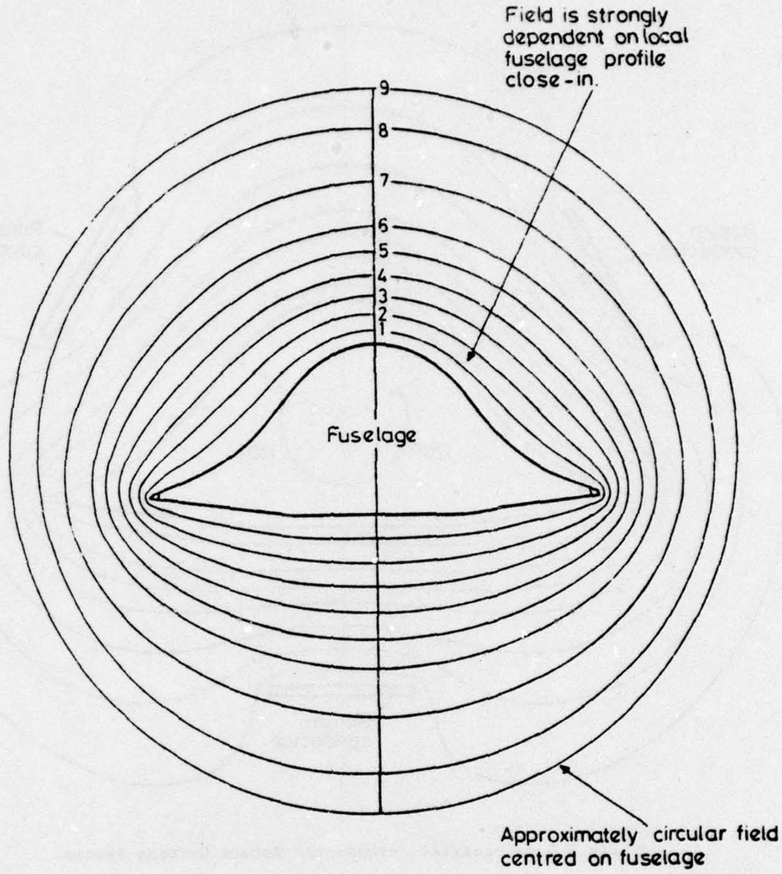

Fig.1 Free Space Field Configuration for Metal Fuselage.

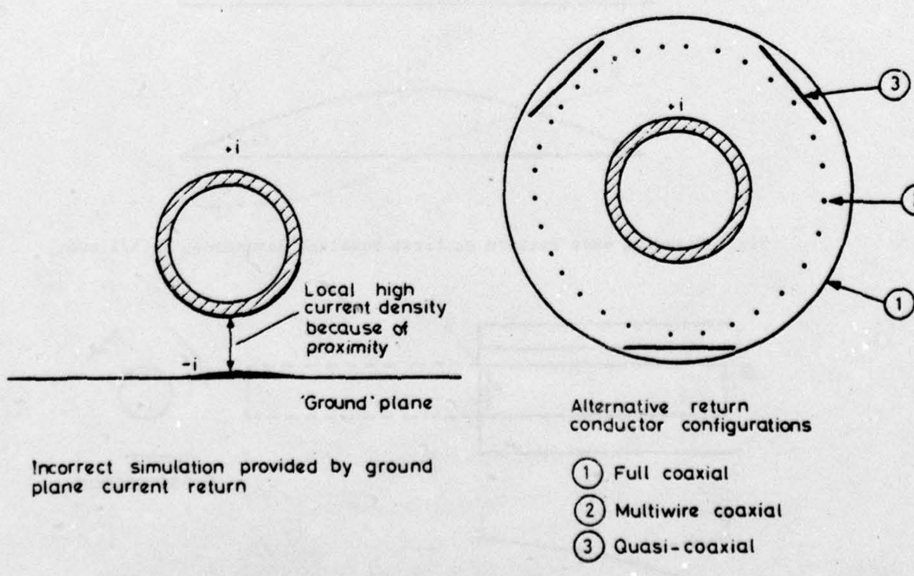

Fig.2 Lightning Simulation on Laboratory Scale.

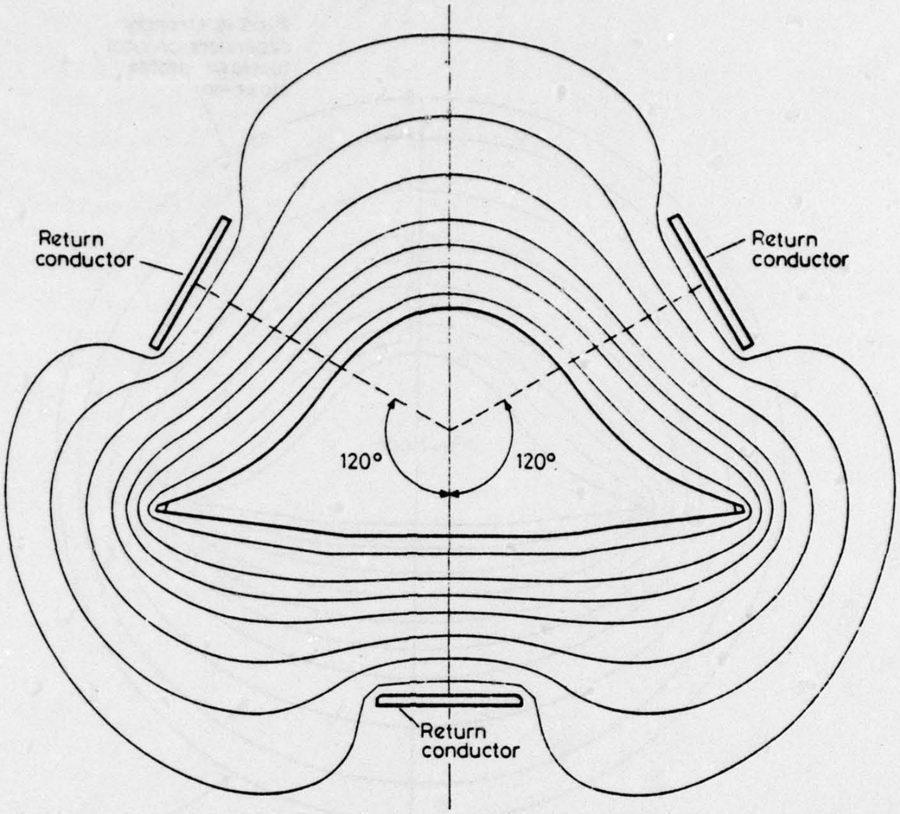

Fig.3 Quasi-coaxial 3-conductor Return Current System.

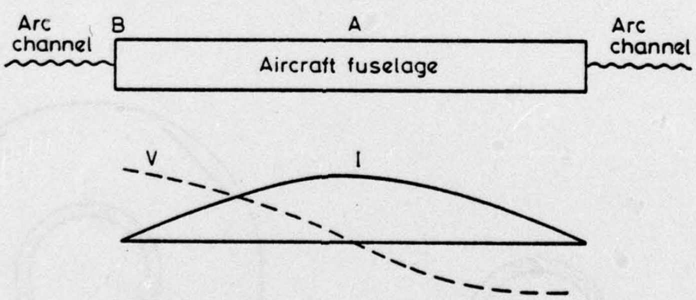

Fig.4 Standing Wave Pattern at First Fuselage Resonance, f_1 ($\lambda/2$ mode)

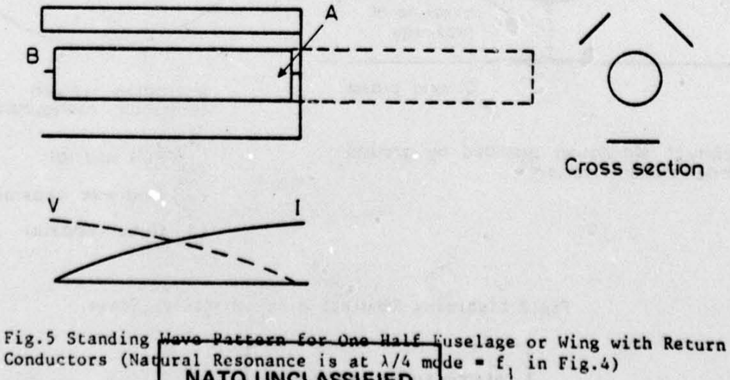

Fig.5 Standing Wave Pattern for One Half Fuselage or Wing with Return Conductors (Natural Resonance is at $\lambda/4$ mode = f_1 in Fig.4)

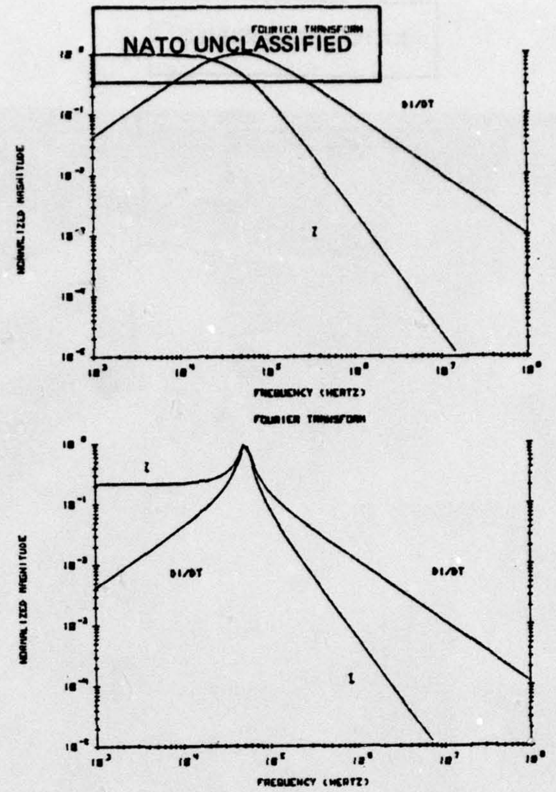

Fig.6 Spectra of Unipolar Pulse (upper) and Damped Oscillatory Pulse (lower).

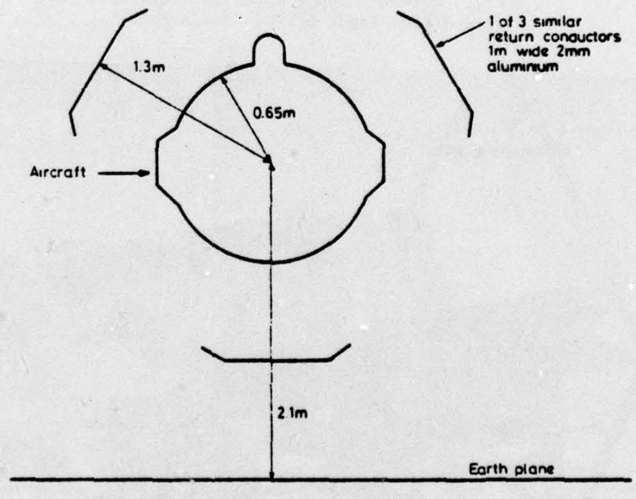

Fig.7 Cross Section A of C.Q.C. Load Assembly.

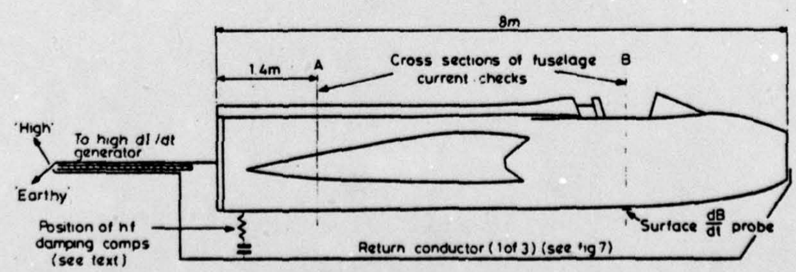

Fig.8 C.Q.C. Load Assembly, side view (only one Return Conductor shown).

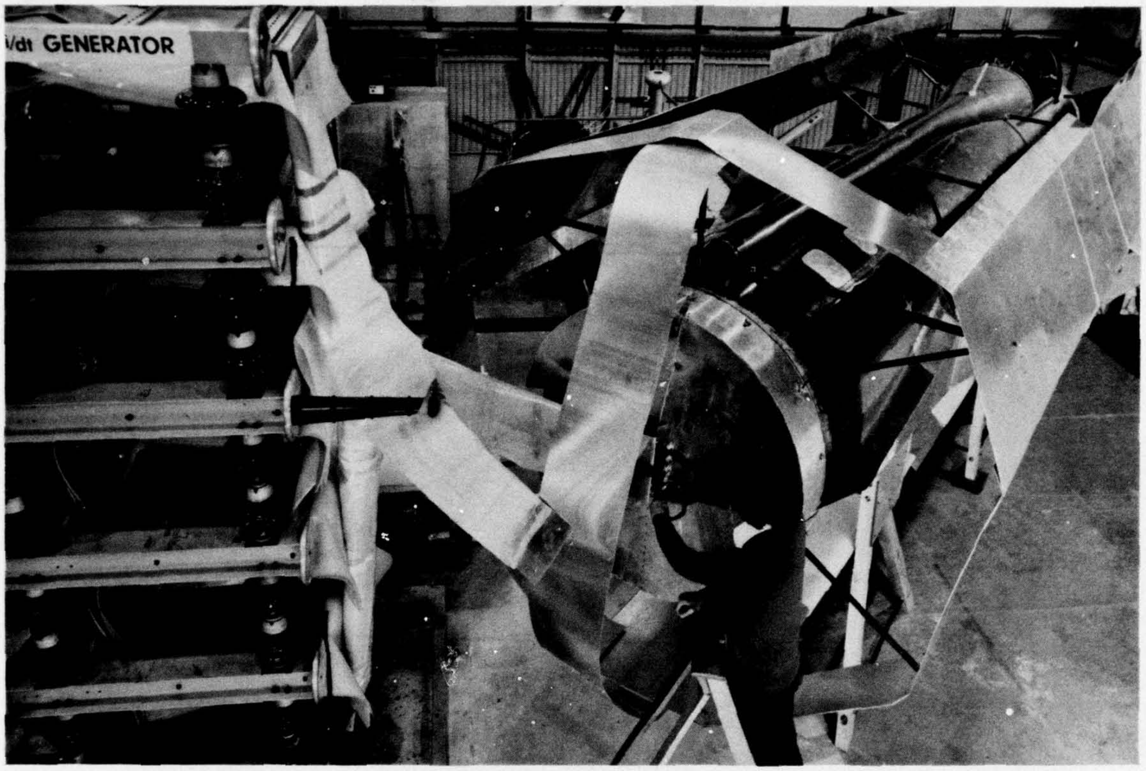

Fig.9 Culham Quasi-coaxial Test System (C.Q.C.) viewed from Input End of Load Assembly.

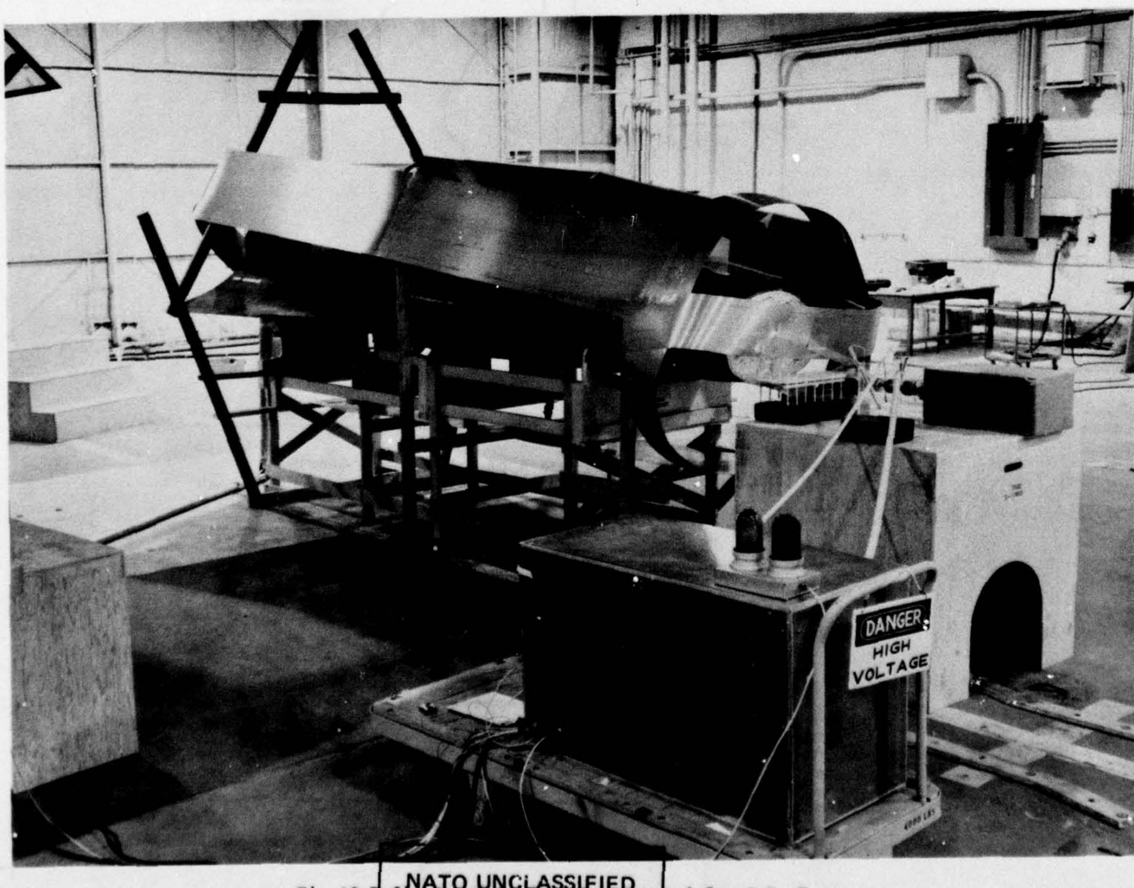

Fig.10 Refined C.Q.C. Test System used for G.D. Tests.

QUESTIONS and ANSWERS

– From D. Clifford

Q – The tests you both describe deal entirely with the effects of current flow through the aircraft. The response measured on the wiring seems to be primarily high frequency in nature, bearing no correlation to either the di/dt or IR response described by B.J.C. Burrows earlier. In view of the fact that Dr Nanevicz reported that HF transients were produced by nearby lightning activity comparable to the transient induced by direct attachment, one might question whether current flow is a factor at all in the production of the HF response. How do you treat circuit responses which have no apparent relationship to peak current or di/dt ?

A – It should be borne in mind that the test techniques described are for an engineering test - not a qualification test. Techniques are under development, and more particularly, data is not available on pre-strike and between-strike fields in a form that can be used for simulation. The test techniques I described allow good simulation of the current pulse through the aircraft and this by itself will produce H.F. transients. Moreover, the current pulse can be shown to produce large voltages in structures incorporating significant amounts of glass fibre reinforced practice or carbon fibre reinforced plastic. Effects due to the current flow are particularly important for carbon fibre structures.

– Comment from R. Hess

The waveform shown by B. Burrows was a result of qualification test and represents the results of directing test arcs to a specific circuit, whereas the waveform shown by J. Corbin represents a circuit response when the complete aircraft is excited. The complete vehicle test represents a situation where the distribution of the test energy internal to the aircraft is not directed but is measured for engineering information and is in the category of engineering tests. The general area of the mechanism controlling the distribution of lightning energies (direct strikes near fields and pre-strike activity) are still under investigation. As a matter of fact projecting from some of the waveforms induced on the internal sensor (length of wire) in the Lear jet used for flight test measurements of lightning activities, it seems to be evident that there are electromagnetic waves with submicrosecond rise times being generated during pre-strike lightning activity.

VULNERABILITY ASSESSMENT OF AIRCRAFT SYSTEMS TO INDIRECT LIGHTNING EFFECTS

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SUMMARY

This paper discusses the factors that constitute the most important considerations in conducting a vulnerability assessment of aircraft systems to indirect lightning effects. It begins with a brief introduction of the characteristics of lightning, its dual hazard to aircraft systems, and its increasing hazard to electronic systems. A general approach and a modified approach to vulnerability assessment are suggested in which a combined analysis and test program is recommended and hardening considerations explored. Analysis is discussed in terms of an electromagnetic coupling model similar to that used for the nuclear electromagnetic pulse (NEMP). The lightning source term and its analytic model are compared in both the time and frequency domains. Test requirements for indirect effects testing of a total vehicle which are based upon SAE recommended practices are reviewed. A linear transient analysis (LTA) test based upon the above requirement is described and improvements in its application are noted. Shortcomings of present tests in terms of simulating the natural environment are mentioned. The usefulness of an analytical model for prioritizing test measurements is noted. Another low level test, continuous wave (cw) swept frequency testing, is described and its usefulness in verification and development of electromagnetic coupling models noted. Finally, circuit and component testing are discussed and an example of the approach to determine component susceptibility to lightning in a given aircraft electronic system is examined. Considerations as to needed improvements in indirect effects testing and improved modeling are given. Facilities for evaluating the vulnerability of the entire aircraft system to realistic lightning threat levels are recommended.

1. INTRODUCTION

Lightning is the most serious natural electromagnetic hazard to aircraft. The occurrence of lightning and its electrical parameters are outlined in Table 1 (Ref. 1).

Lightning can cause a variety of direct and indirect effects on aircraft if struck. In addition, precipitation static electrification effects produced by the flight of an aircraft through ice, snow, or rain can generate potentials exceeding 50kV and cause hazards ranging from minor to catastrophic. A description of these threats and an illustration of systems susceptible to both lightning and static electrification effects are given in Table 2 and Figure 1, respectively (Ref. 2).

Lightning is a dual hazard to aircraft because of both its damaging direct, or physical, effects as well as its induced or indirect electromagnetic effects. The most obvious direct effects are physical damage and burnout of the aircraft structure caused by extreme heat loading as a result of high current lightning pulses. Resultant pitting/puncture points indicate the location of lightning entry/exist points on the aircraft. The most frequent direct effect damage occurs at radomes, pitot booms, canopies, external antennas, and wing tips. Unless protective devices and designs are used, damaging currents can penetrate directly into the interior of the aircraft.

Indirect or induced effects due to lightning may cause temporary disruption (upset) or permanent damage to internal electrical/electronic systems. Lightning currents on the skin of the aircraft (which result from a direct strike or a nearby strike) can couple energy into the aircraft interior to produce voltage and current surges on sensitive electrical circuits. The coupled energy flow path is shown in Figure 2 (Ref. 3). Important lightning parameters include rise time of the current pulses, peak current, total charge transferred, peak electric and magnetic fields, and the frequency spectrum of radiated fields.

Because earlier vacuum tube electronics were relatively immune to transients or surges induced in aircraft electrical circuits by lightning or static discharges, induced effects were of little concern. However, during the past decade, electronics technology has progressed to discrete solid state electronics and more recently to integrated circuit electronics. The next step, which is now underway, is the application of microcircuitry including largescale integrated circuits, microprocessors, memories, and similar devices in new commercial and military aircraft and in new missile systems. These components are inherently more susceptible to voltage and current surges because of their low operating voltage, current, and power handling capabilities. Moreover, they are being used in critical electronic subsystems which may be essential for the integrity of safe and controlled flight. Even use of circuit and component "redundancy" may not prevent interruption or failure of a critical subsystem caused by simultaneous lightning-induced transients. Evidence of impact or damage to avionic/electrical systems without direct lightning attachment to electrical components has recently appeared in airline lightning strike reports (Ref 4). At present, there are no adequate standards, specifications, or criteria specifically applicable to protection against these effects.

Another factor that is contributing to awareness of induced effects is the increasing trend for greater use of advanced composite materials in aircraft structures. These composite materials (e.g., graphite epoxy) have inherently reduced electromagnetic shielding properties over the frequency spectrum generated by lightning. Hence, sensitive electronic flight controls and avionics may be exposed to considerably higher levels of induced electromagnetic energy if proper protective design and installation techniques are not employed.

2. APPROACHES TO VULNERABILITY ASSESSMENT FOR INDIRECT EFFECTS

Vulnerability assessments of aircraft systems to a lightning environment can be made by using test, analysis, or a combination of both. Maximum confidence in the assessment is obtained by the coordinated use of both test and analysis. Tests are required to verify submodels of the system under assessment.

2.1 General Approach

Figure 3 shows a general approach to be taken in assessing the vulnerability of aircraft systems to the indirect effects of lightning and in applying (as needed) the selective integration of hardening technologies for flight and mission critical systems (Ref. 5).

A lightning source description for either direct attached or nearby lightning (Block 1) is developed and becomes the source term for direct energy to the aircraft. An aircraft electromagnetic (EM) coupling model (Block 2) is developed using aircraft configuration data (Block 3) and results of low-level direct-drive ground testing (Block 4).

Lightning induced transients that are predicted by the EM coupling model are compared to calculated system upset and damage thresholds (Block 5) to determine the safety margin at the circuit level (Block 6). The safety margin is a measure of the vulnerability and is defined as the difference (in dB) between the circuit threshold and the amplitude of the induced transient at the input. Its interpretation must take into account the data quality associated with the techniques used to determine both the induced responses and the circuit thresholds. If the safety margin is negative or judged to be inadequate, then hardening is required.

Hardening requirements are initially developed and then hardening options are identified. These options may include hardening at the penetrations (e.g., windshield, wing wires, door slots, control cables), at the internal cabling, at the equipment rack, or in the equipment design itself (Block 7). Each of these hardening options is evaluated for its impact on cost, system performance, reliability and maintenance, as well as its electrical performance as a hardening measure/device.

Ultimately, trade studies define a mix of hardening measures/devices that are incorporated into a modified EM coupling model (Block 8). The hardening measures/devices are then physically implemented and low-level tests (Block 9) are conducted on the hardened aircraft (Block 10) to validate the hardened design. This validation is achieved through examination of safety margins again (Block 12) and threshold data obtained on the critical circuitry used in the subsystem equipments (Block 11). The product of analysis and low-level testing is a validated lightning-hard design for a particular set of aircraft systems and a particular airframe configuration.

For final verification of the hardened aircraft, a series of lightning simulation tests is conducted (Block 13). The tests require extensive simulation capabilities, extensive monitoring of critical circuits, and a sophisticated data handling system to record and reduce experimental measurements and to compare the results with predicted responses.

The general approach is applicable for retrofit hardening of existing aircraft or missiles. However, if it is not possible to describe the aircraft configuration in sufficient detail to develop an accurate EM coupling model, a modified approach can be taken.

2.2 Modified Approach

A modified approach may have to be taken when new systems are being developed and an airframe has not yet been selected in which to house the system. It is still possible to assess the vulnerability of the new system to lightning induced effects and to evaluate a mix of hardening trade-offs on a selective basis.

The first step is to perform a subsystem threshold analysis to determine the inherent failure threshold in a lightning environment. The analysis is conducted to identify those circuits potentially susceptible to permanent damage and transient upset and to determine if potentially susceptible circuits are flight or mission critical.

Because an aircraft EM coupling model has not been developed, lightning induced transients cannot be predicted at the circuit level. Hence, the vulnerability of the susceptible circuits cannot be established with the same degree of confidence as was accomplished in the general approach.

Instead of using predicted lightning induced transient levels, arbitrary levels are established that bound the most probable induced transient level. These levels are compared with selected system circuit thresholds and arbitrary safety margins are determined. If the safety margins are negative, then hardening is required.

Hardening requirements are developed and hardening options are identified as in the general approach. Because a multiplicity of hardening levels may result from establishing arbitrary transient input levels, a more extensive trade-off analysis may be required to adequately assess the increased number of options. Again, trade studies define a mix of hardening measures/devices to use based on hardening requirements.

Circuit threshold tests are performed based on the initial subsystem threshold analysis to obtain validated threshold data.

The final product is (1) a well-defined set of hardening measures/devices to apply for arbitrary negative levels of safety margin and (2) a complete set of threshold data for selected subsystem circuits.

As soon as a prototype airframe is chosen to house the new system, the procedures in the general approach can be taken to obtain a validated lightning-hard design by using the applicable information obtained from hardening trade analyses and circuit threshold tests.

3. ELECTROMAGNETIC COUPLING MODEL

The usual approach in electromagnetic modeling is to use frequency domain transfer functions for the separate coupling mechanisms. A spectral representation of the threat environment and the test driving source is also required. The coupling mechanisms are generally classified as external models, penetration models, and internal models. The external model is a transfer function of a surface current density and charge on the skin of the aircraft due to a unit amplitude source. The penetration models are transfer functions relating surface current and charge on the skin to internal quantities such as cavity fields or induced voltages on internal cables. The final transfer function to the critical element being assessed is usually a transmission line model of the aircraft wiring which relates the induced cable voltage to a pin voltage at the electronic equipment (Ref. 6 and 7).

The techniques used to model electronic systems in a lightning environment are virtually identicial to those developed over the past 10 to 15 years for nuclear electromagnetic pulse (NEMP) assessment once the external skin responses are determined (Ref. 8-14).

The source term for directly attached lightning and a nearby lightning stroke is usually approximated by a unidirectional current pulse of the form

$$i_L(t) = A(e^{-\alpha t} - e^{-\beta t}) \qquad (1)$$

The lightning current often assumed is the severe return stroke model developed by Cianos and Pierce (Ref. 15). It is a double exponential current with a peak amplitude of 200 kiloamperes, a zero to peak time of 1.5 microseconds, and a time to half peak of 40 microseconds. The time history of this current is shown in Figure 4.

For nearly lightning, the fields produced by the stroke have been approximated as those due to an infinite current column with the same current waveform as that for direct attachment.

In the frequency domain, there exist three frequency regions of interest for $\beta >> \alpha > 0$ as shown in Figure 5. The upper turning frequency, β, is considerably below the first resonance of most aircraft.

Although this unidirectional pulse waveform has been used for a number of years, many of the features of natural lightning are not incorporated in the waveform. A more representative model is a triple exponential of the form

$$i_L(t) = A(1 - e^{-\gamma t})(e^{-\alpha t} - e^{-\beta t}) \qquad (2)$$

where

$$\gamma >> \beta >> \alpha > 0$$

It starts in a physically more credible manner with zero slope at the starting time. In the frequency domain, there is a third turning point with the consequence that the higher frequency amplitudes are decreased. Figure 6 shows a comparison of the double and triple exponential representation of the lightning pulse in the time and frequency domains.

4. TEST REQUIREMENTS

The primary test requirement is to simulate the mode or modes of excitation of the aircraft which best represent the actual threat environment. Since it is not practical to try to duplicate a full scale lightning strike with currents up to 200 kA and because of the potentially destructive nature of such currents, lightning simulation testing has been conducted at reduced current levels using either the unidirectional (double exponential) or the damped sinusoidal waveform.

4.1 Indirect Effects Testing

An outline for conducting indirect effects testing of a comple vehicle is given in Section 4.2.3 of Reference 16 and is reproduced here for convenient reference. A

description of test waveforms given in Section 3.3.3 is also reproduced.

4.2.3 Indirect Effects - Complete Vehicle

4.2.3.1 Objective

The objective of this test is to measure induced voltages and currents in electrical wiring within a complete vehicle. Complete vehicle tests are intended primarily to identify circuits which may be susceptible to lightning induced effects.

4.2.3.2 Waveforms

Two techniques, utilizing different waveforms, may be utilized to perform this test. One involves application of a scaled down unidirectional waveform representative of a natural lightning stroke.

The second technique involves performance of the test with two or more damped oscillatory current waveforms, one of which (component G_2) provides the fast rate of rise characteristic of a natural lightning stroke wavefront, and the other (component G_1) provides a long duration period characteristic of natural lightning stroke duration. Induced voltages should be measured in the aircraft circuits when exposed to both waveforms and the highest induced voltages taken as the test results.

Each test is carried out by passing test currents through to the complete vehicle and measuring the duced voltages and currents. Checks are also made of aircraft systems and equipment operations where possible.

4.2.3.2.1 Unidirectional Test Waveform

Waveform F should be applied.

4.2.3.2.2 Oscillatory Waveforms

Waveforms G_1 and G_2 should be applied.

4.2.3.3 Test Setup

The test current should be applied between several representative pairs of attachment points such as nose-to-tail or wing tip-to-wing tip. Typical test setups are shown in Figure 4-7.

Attachment pairs are normally selected so as to direct current through the parts of the vehicle where circuits of interest are located.

Multiple return conductors should be used to minimize test circuit inductance and proximity effects. Typical test setups are shown in Figure 4-7.

4.2.3.4 Measurements and Data Requirements

The test current amplitude, waveform, and resulting induced voltages and currents in the aircraft electrical and avionics systems should be measured.

CAUTION: Interference-free operation of the voltage measurement system should be verified.

Voltages measured during the complete vehicle tests should be extrapolated to full threat levels in the same manner as described in Para. 4.1.6.2 for indirect effects measurements in external electrical hardware. Situations such as arcing paths or nonlinear impedances exist which may result in nonlinear relationships between induced voltages and applied current. Careful study of the vehicle under test, however, can usually identify such situations. When testing fueled vehicles, care should be taken to prevent sparks across filler caps, as even low amplitude currents can cause sparking across poor bonds or joints. In doubtful situations, fuel tanks should be rendered nonflammable by nitrogen inerting.

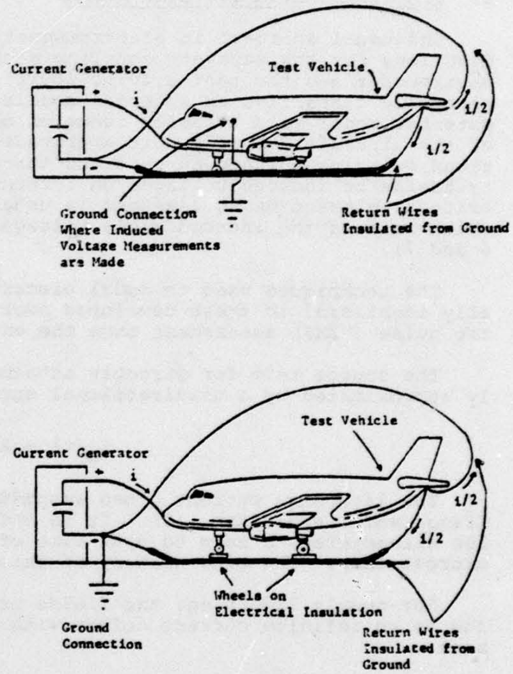

Figure 4-7. Typical setups for complete vehicle tests.

3.3.3 Current Waveforms

Current waveform components F and G, shown on Figure 3-5, are intended to determine indirect effects on very large hardware and full size vehicles. These waveforms are specified at reduced amplitudes to overcome inherent full vehicle test circuit limitations and also to allow testing at non-destructive levels to be made on operational vehicles at non-destructive levels. Scaling will depend on the nature of the coupling process as detailed in the following paragraphs.

3.3.3.1 Test Waveform F - Reduced Amplitude Unidirectional Waveform

Component F simulates, at a low current level, both the rise time and decay time of the return stroke current peak of the lightning flash. It has a rise time of 2 microseconds (±20%), a decay time to half amplitude of 50 microseconds (±20%) and a minimum amplitude of 250 amperes. Indirect effects measurements made with this component must be extrapolated to the full lightning current amplitude of 200 kA.

3.3.3.2 Test Waveforms G_1 and G_2 - Damped Oscillatory Waveforms

Fast rate of rise current waveforms and higher amplitude waveforms may often be usefully employed for indirect effects testing. For indirect effects dependent upon resistive or diffusion flux effects (i.e. not aperture coupling) a low frequency oscillatory current - waveform G_1, in which the period (1/f) is long compared with the diffusion time, should be used. This requires a frequency, f, of 2.5 kilohertz or lower (i.e. the duration of each half-cycle is equal to or greater than 200 μs). Where resistive or diffusion effects are measured, the scaling should be in terms of the peak current, with full scale being 200 kA.

For indirect effects dependent upon aperture coupling the high frequency current, waveform G_2, should be used. The maximum frequency of waveform G_2 should be no higher than approximately 300 Khz or 1/10 of the lowest natural resonant frequency of the aircraft/return circuit, whichever is lower. Where aperture-coupled effects are measured the scaling should be in terms of rate-of-rise (di/dt), with full scale being 100 kA/μs.

When testing composite structures with waveform G_2, resistive and diffusion flux induced voltages may occur as well as aperture coupled voltages, and results should be scaled both to 200 kA and to 100 kA/μs.

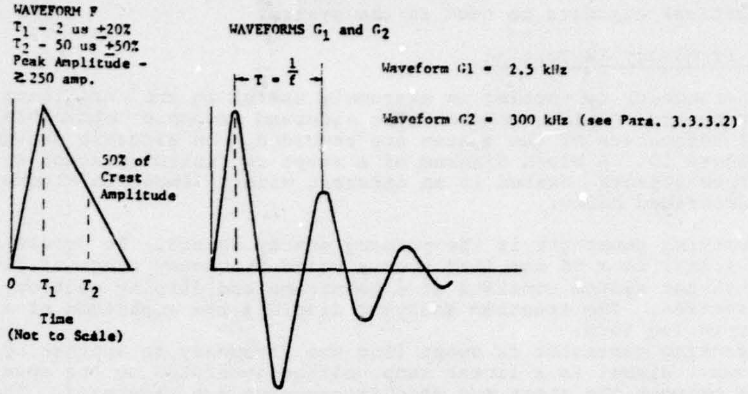

Figure 3-5 Idealized current waveforms for engineering tests. (Note: Peak amplitudes are not the same.)

The indirect effects test described above is commonly called the lightning transient analysis (LTA) test and was first developed by the General Electric Company High Voltage Laboratory under sponsorship of the NASA Aerospace Safety Research and Data Institute (Ref 17). LTA tests have been conducted on a variety of flight vehicles. General Electric has applied the test to the NASA Flight Research Center F-8 digital fly-by-wire airplane, the Navy-Lockheed S-3A antisubmarine warfare aircraft, and the Air Force-General Dynamics YF-16 air combat fighter. The Air Force Flight Dynamics Laboratory has conducted LTA tests on F-104, F-4, YF-16, A-7, and FB-111 aircrafts during the past three years (Ref 18).

A number of improvements have been made in the test technique in recent years (Ref 19). A typical test configuration is shown in Figure 7. To minimize circuit inductance and distributed capacitance, multiple parallel return leads are used and are supported on non-metallic stanchions. The symmetry of the configuration tends to balance the currents flowing in the fuselage and to minimize the influence of the return path. A ten-fold increase in the current output of the generating equipment has been made. The generator can supply up to 3,000 amperes to full scale fighter aircraft. It consists of two separate banks of parallel capacitors charged simultaneously using a 100kV dc power supply and is discharged in series when a desired voltage is reached. Pneumatically actuated devices have replaced electronic triggering shorting devices to eliminate critical gap adjustments and to reduce rf interference. The approach has the potential of a further ten-fold increase in current by adding additional capacitor banks.

Measurement system isolation is achieved using small breakout boxes and special cables. The breakout boxes are connected to the aircraft circuit with short multi-conductor insulated wire bundles inside a braided copper shield. The wire pair selected is connected to an electronic system that converts the induced signal to a modulated light beam which is transmitted via a fiber optic cable to a receiver for reconversion to an electrical signal. Complete electrical isolation is achieved between the test circuit and the data acquisition system. The applied current is measured using a current monitoring transformer that is electrically isolated from the current generator.

Initially, the data acquisition system consisted of coaxial (or twin axial) cable inputs to an oscilloscope. The transient measured was recorded photographically. The resulting oscillogram was manually converted to tabular data which could be digitized for computer processing and analysis. Today, the oscilloscope is replaced by a signal processing system consisting of a transient digitizer, minicomputer, CRT display, graphic terminal, cassette and/or floppy disk drives and a hard copy printout unit. The CRT displays the transient signal while the transient digitizer digitizes and stores it. Using built-in software, the minicomputer can process the data and the processed data can be displayed on the graphic terminal. The data can be further manipulated through instructions via a keyboard to the graphic terminal and stored on a tape cassette or floppy disk. The built-in software includes plotting routines and Fast Fourier Transformations (FFT's). The hard copy printout device provides a permanent record of any of the stored or displayed data as illustrated in Figures 8 and 9.

One of the shortcomings of the present indirect effects testing is that it produces very little electric field. A combination of a high impedance-high electric field (low current) simulator in conjunction with a low impedance-high current simulator would increase the external induced electric field environment which corresponds to corona or streamer production from vehicle extremities.

An analytical model is useful in setting up the priorities of the test measurements. For example, many of the measurements will have similar responses since they are on the same electrical cable and are excited by the same penetration. Pretest predictions help in eliminating many of the redundant measurements so that a broader sample of responses is obtained. Measurements may be prioritized based upon the maximum voltage expected.

Close coordination with circuit designers is required for selecting and prioritizing the most critical circuits to test in the system.

4.2 Swept Frequency CW Testing

Swept frequency cw testing is extremely useful in the verification and development of electromagnetic coupling models. The wideband response obtainable assures that all significant resonances of the system are recorded. An aircraft set-up for cw testing is shown in Figure 10. A block diagram of a swept cw instrumentation system used to record responses from sensors located in an aircraft wing is shown in Figure 11 (Ref. 20). The system is described below.

The tracking generator is the primary energy source. It provides a constant amplitude sinusoidal signal to a 50 ohm load over a broad frequency range of 100 kHz to 110 MHz. The spectrum analyser system consists of a mainframe and display section, an IF section, and an rf tuning section. The spectrum analyser displays the amplitude of a signal as a function of frequency in log form.

The tracking generator is swept from one frequency to another by the spectrum analyzer. The swept signal is a linear ramp voltage generated by the spectrum analyzer. All frequencies between the start and stop frequencies are generated. The instantaneous frequency of the tracking generator output signal is phase-locked to the tuning frequency of the spectrum analyzer so that the frequency deviation between the two is less than one hertz. This permits the spectrum analyzer to measure very small signals by using a narrow band detector. Typically, signal levels less than -120 dBm can be detected by the spectrum analyzer in a 100 Hz bandwidth.

A wideband rf power amplifier boosts the amplitude of the signal applied to the aircraft thereby increasing the dynamic range available for the measurements. The output of the amplifier is also swept cw but varies in amplitude because the impedance of the aircraft varies with frequency. The maximum output signal was one ampere current and three watts power.

The outputs from magnetic field, electric field, and current sensors installed at various locations on the wing are transmitted to the spectrum analyzer by a fiber optics amplifier link. The fiber optics system consists of a transmitter, a fiber optics cable, and a receiver. The fiber optics system isolates the sensor from the spectrum analyzer and prevents test cable interaction. It reduced the measurement sensitivity to -100 dBm. The transmitter was designed with a differential, high impedance input circuit so that the fiber optics system could be used as a voltage probe.

To protect test equipment from stray electromagnetic fields produced by the aircraft acting as a transmission line, the tracking generator, spectrum analyzer, and fiber optics receiver were placed inside a shielded box and an rf power line filter was installed at the shielded box to reduce rf current penetration.

4.3 Circuit and Component Testing

The objective of circuit and component testing is to establish a data base of computer failure or circuit upset or to verify design margins. Because of the objectives, testing does not require a rigorous reproduction of lightning induced current or voltage. In fact, the published data base for semiconductor damage constants was obtained using a pulse having a rectangular waveform. In turn, this can be simply related to a pulse having a sinusoidal waveform.

A typical set of components susceptibility test criteria that was developed and then followed to determine the hardness of a particular aircraft electronics system to lightning induced effects is given below.

(1) The interface circuits of the system were reduced to a minimum number of worst-case generic forms.

(2) Each generic test circuit was subjected to four unidirectional rectangular pulses in the worst-case direction.

(3) The test pulse had an open circuit voltage of at least 500 volts and a source impedance of less than 100 ohms.

(4) The test pulse width was at least 5 microseconds in duration with a rise and fall time not greater than 15 percent of the duration.

(5) In general, at least four samples of each generic test circuit containing solid state components were tested.

(6) Generic test circuits that failed were evaluated for hardening techniques.

(7) If feasible, the failure level for each circuit was determined by increasing the pulse amplitude or decreasing the source impedance.

The 500 volt pulse was chosen because it represented the maximum extrapolated peak voltage measured using the LTA test. The peak voltage measured represented negligible energy in comparison to the 500 volt, 5 microsecond pulse. The 5 microsecond pulse width represented the maximum time that any significant transient voltage was measured using the LTA test.

A high power pulse generator and a 100 MHz multi-mode storage oscilloscope were used to conduct the tests. Two different impedance matching plug-in output units for the pulse generator were used for the different circuit arrangements. The first test arrangement, shown in Figure 12, was used to couple relatively low, nonlinear impedance test circuits. The pulse generator load for this circuit was relatively insensitive to test circuit impedance so that the test circuit could be subjected to a 600 volt pulse through an 82 ohm limiting resistor (lightning "source impedance"). The second test arrangement, shown in Figure 13, was used for testing relatively high impedance test circuits for voltages above the criteria level. This test arrangement made the load on the pulse generator relatively insensitive to test circuit impedance when the test circuit impedance was significantly greater than 50 ohms. This arrangement allowed application of voltages twice the pulse amplitude for many of the test circuits.

5. CONCLUSIONS AND RECOMMENDATIONS

This paper has discussed methods and techniques that are presently used or recommended to assess the vulnerability of aircraft systems to the indirect effects of lightning. Methods of analysis and modeling have been outlined and testing requirements and techniques reviewed. The need for both analysis and testing to be done concurrently to optimize results cannot be too strongly emphasized.

Improvements to more realistically simulate the natural lightning environment when conducting ground-based indirect effects tests need to be considered. Additional in-flight data are needed to improve the lightning/aircraft interaction model and to determine if present waveforms being used are adequate. Eventually, ground-based test facilities that are capable of testing full-scale aircraft to realistic lightning threat levels should be used to evaluate the vulnerability of the total aircraft system.

6. REFERENCES

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TABLE 1 - LIGHTNING CHARACTERISTICS

ELECTRICAL

- TYPES - Intra/intercloud, cloud-ground, positive, negative
- POTENTIAL - 30-100 million volts
- CURRENT - 20-200 thousand amps (peak)
- POWER - 10^{12} watts nominal (peak)
- ENERGY - 5 x 10^{8} joules nominal (200 lb TNT equivalent per stroke)
- EXTENT - 3-30 km/stroke (path is frequently predominately horizontal)
- SPECTRUM - Peak energy near 10 KHz, some above 10 MHz
- DURATION - STROKE - 100 micro sec

  FLASH - 0.2 sec (1-20 strokes)

OCCURRENCE/EFFECTS

- Worldwide phenomenon; 1 flash every 20 sec on average in a storm, 1800 storms simultaneously worldwide; activity varies with climate, season, hour, location, altitude. Turbulence generally correlated with lightning activity.
- Aircraft penetration through high electric field region may trigger lightning strike. Two or more attachment points for each strike.
- Commercial airline data - about one direct strike per aircraft annually, many nearby strikes.
- Air Force data - fewer strikes shown than commercial due to mission profiles, avoidance, reporting procedures. Much greater strike frequency in European Theater due to greater activity and route constraints. During 1968-1977 USAF aircraft dollar value losses averaged $1.5M/year by conservative official estimates.

TABLE 2 - ATMOSPHERIC ELECTRICITY THREATS TO AIRCRAFT

Hazard	Cause	Hazard Criticality
Malfunction/failure of electronic control systems	Low tolerance to electrical transients caused by direct/induced lightning or static electrification effects. May simultaneously affect parallel "redundant" system.	Minor to catastrophic
Fuel tank fire or explosion	Fuel vapor ignition caused by static electricity or lightning effects.	Minor to catastrophic
Loss of engine power	Possible lightning acoustic shock at engine inlet, or electrical transient effects on engine controls.	Minor to catastrophic
Inadvertent release/ignition of external stores	Premature activation caused by lightning or static electrification effects	Serious to catastrophic
Radome, canopy, and windshield damage	Direct lightning strikes; arc discharge caused by static electricity buildup.	Minor to serious
Instrumentation problems/communications, navigation & landing system interference	Transient effects caused by static electricity buildup & direct & nearby lightning strikes.	Minor to catastrophic
Structural damage	Direct lightning attachment to aircraft	Minor to serious
Physiological effects on crew	Flash blindness & distracting or disabling electrical shock caused by direct & nearby lightning strikes.	Minor to catastrophic

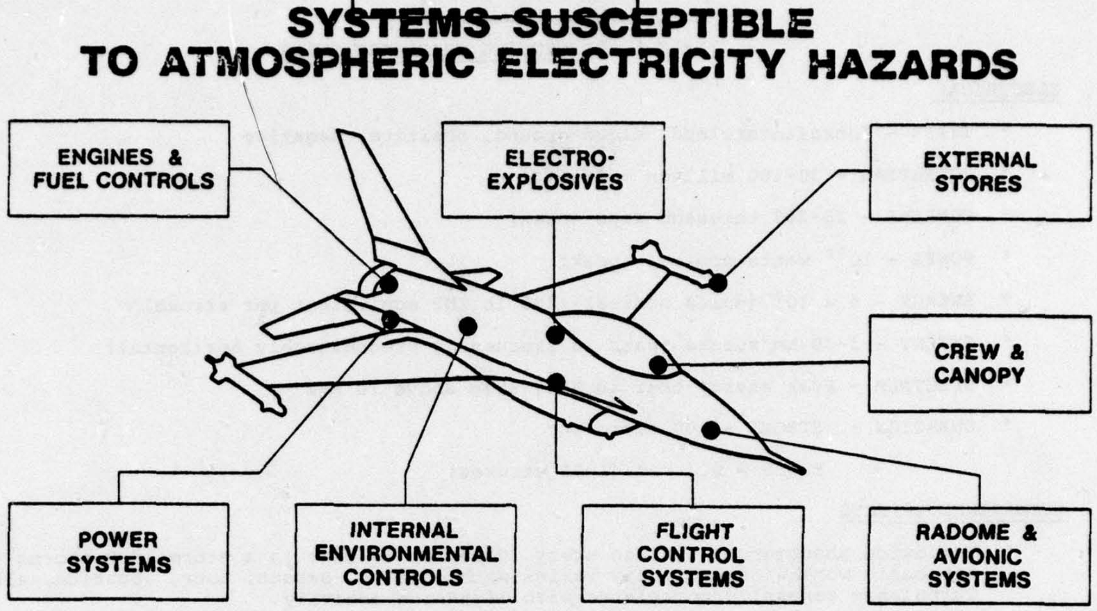

FIGURE 1

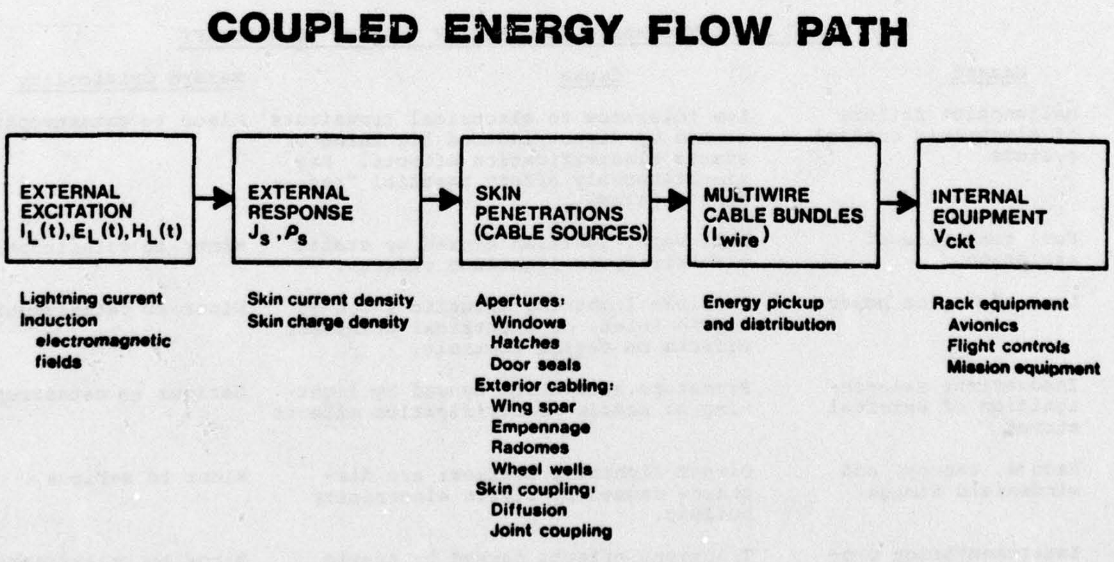

FIGURE 2

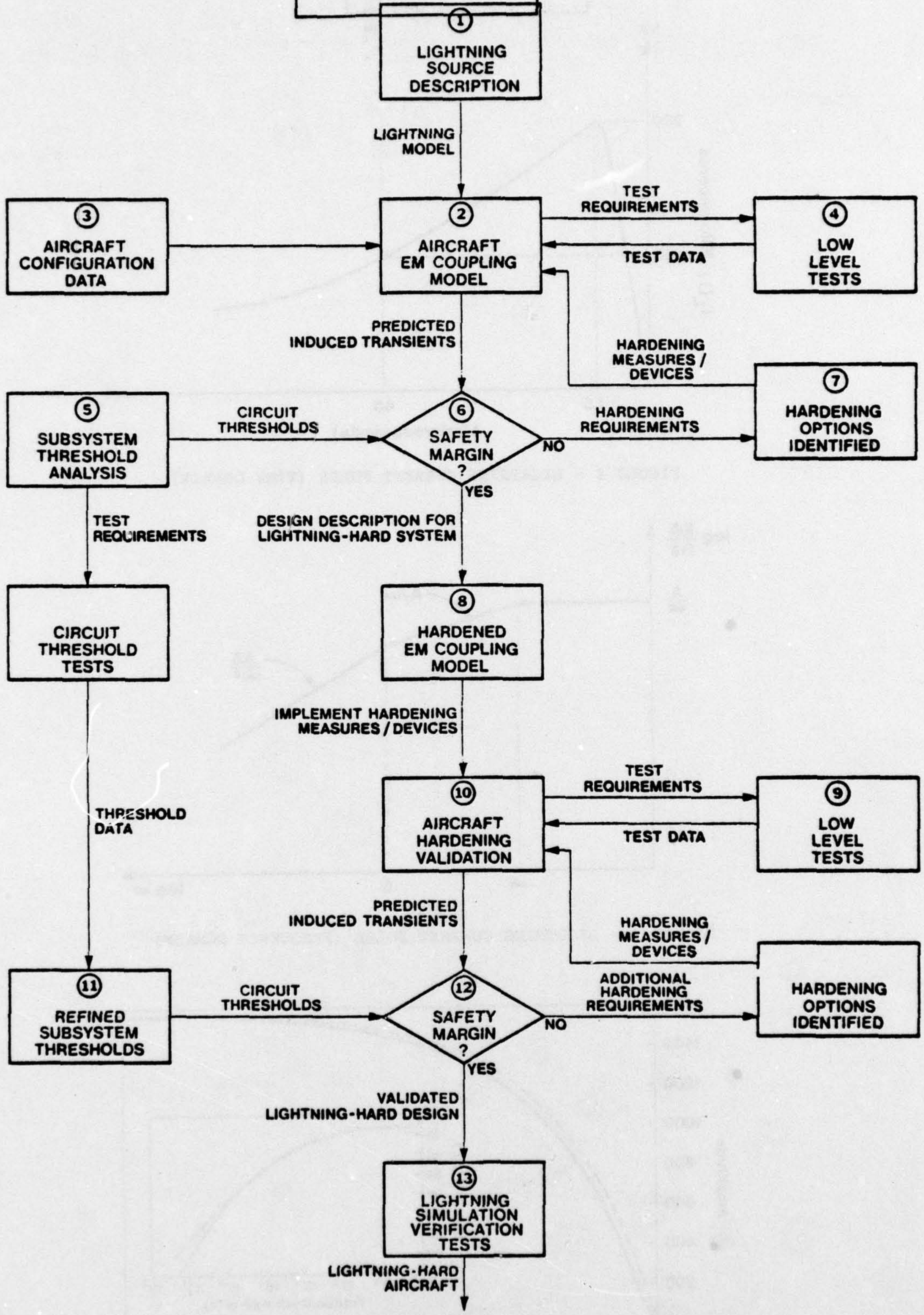

FIGURE 3 - GENERAL APPROACH TO VALIDATED LIGHTNING-HARD AIRCRAFT

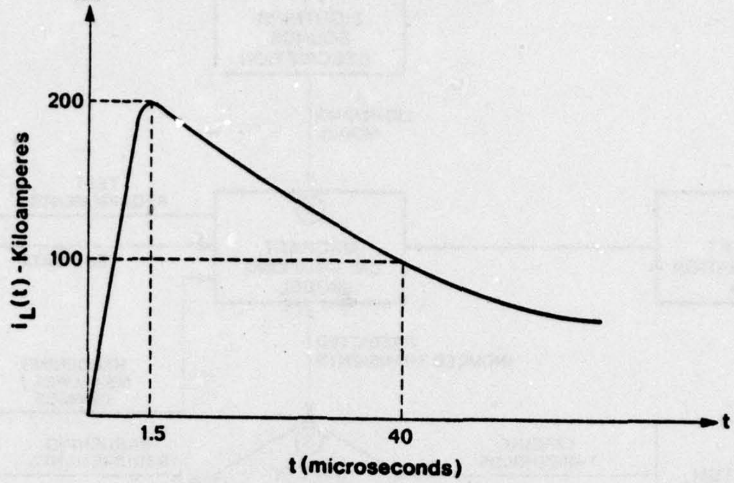

FIGURE 4 - LIGHTNING CURRENT PULSE (TIME DOMAIN)

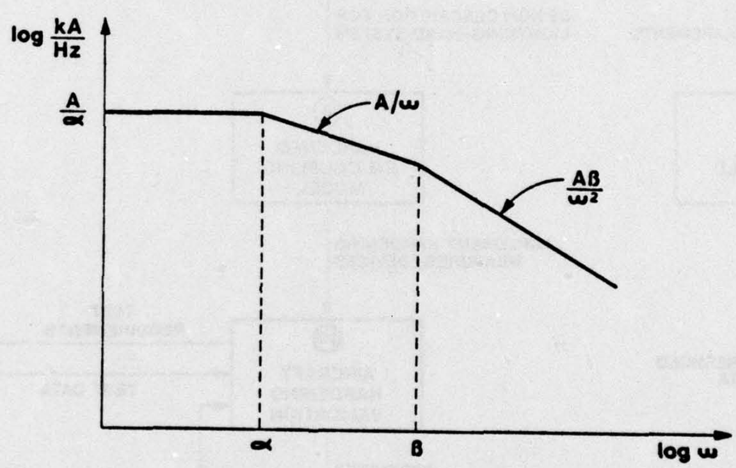

FIGURE 5 - LIGHTNING CURRENT PULSE (FREQUENCY DOMAIN)

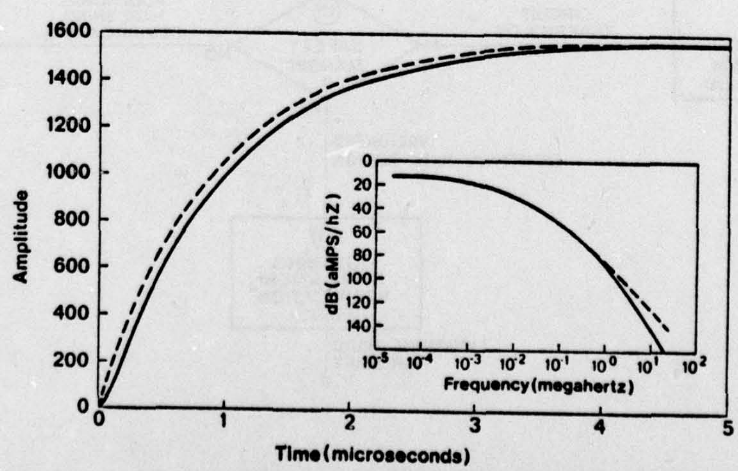

FIGURE 6 - COMPARISON OF DOUBLE AND TRIPLE EXPONENTIAL LIGHTNING PULSE REPRESENTATIONS

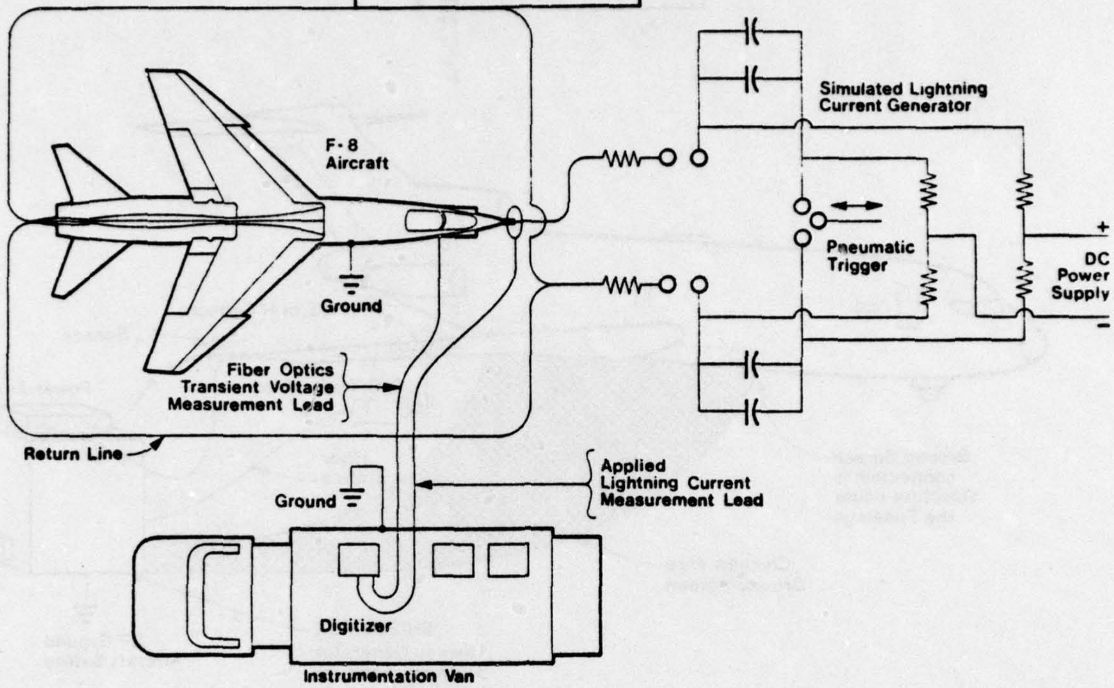

FIGURE 7 - TYPICAL LIGHTNING TRANSIENT ANALYSIS (LTA) TEST CONFIGURATION

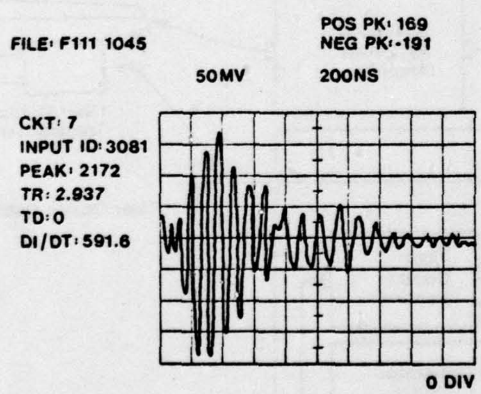

FIGURE 8 - SAMPLE INDUCED TRANSIENT RECORD

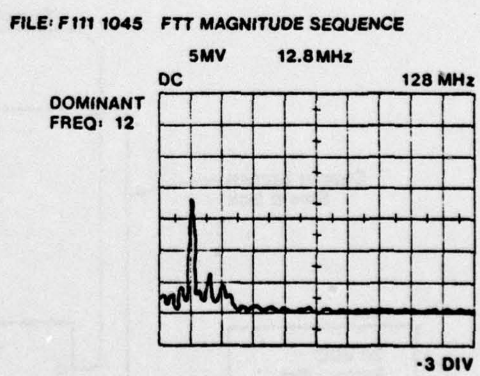

FIGURE 9 - FOURIER TRANSFORM RECORD

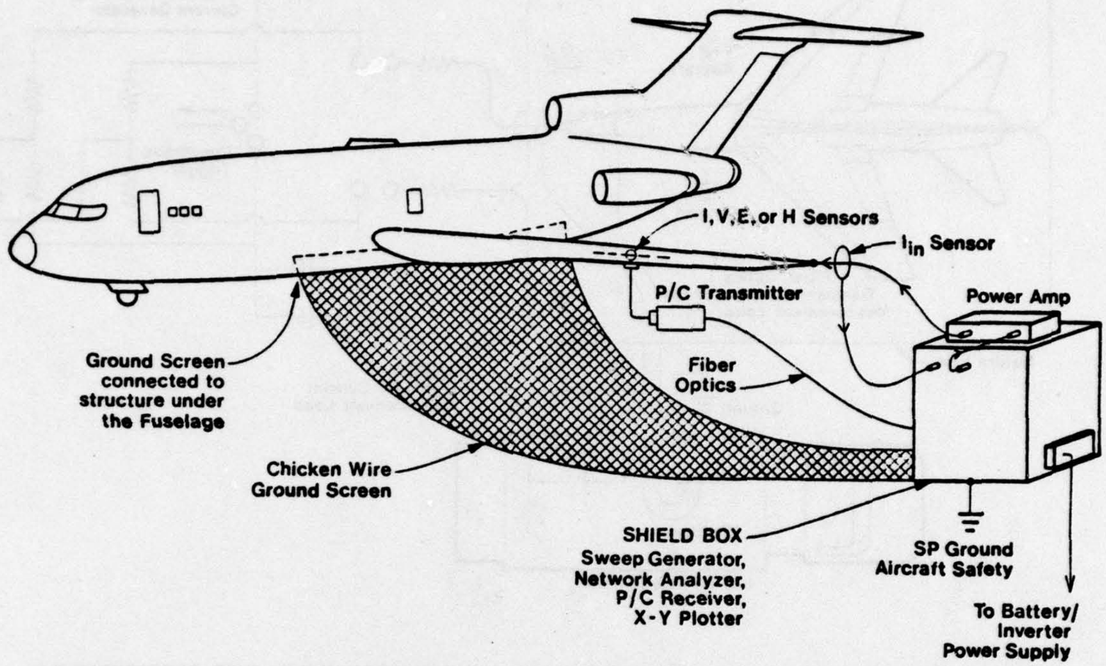

FIGURE 10 - AIRCRAFT CW TEST CONFIGURATION

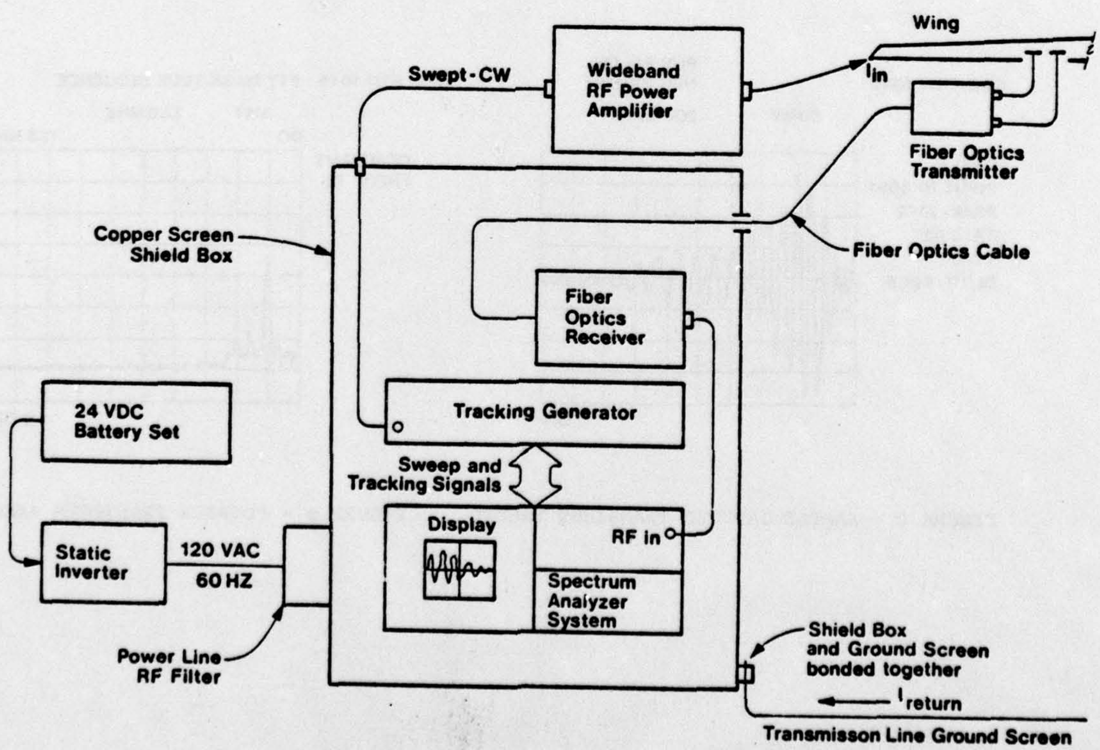

FIGURE 11 - DIAGRAM OF SWEPT CW INSTRUMENTATION SYSTEM

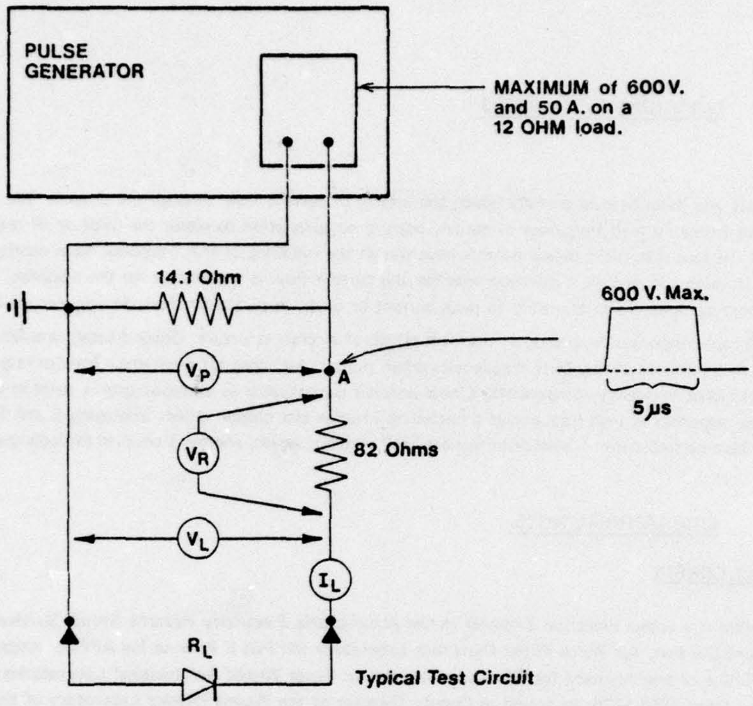

FIGURE 12 - TEST ARRANGEMENT FOR "LOW" NONLINEAR IMPEDANCE TEST CIRCUITS

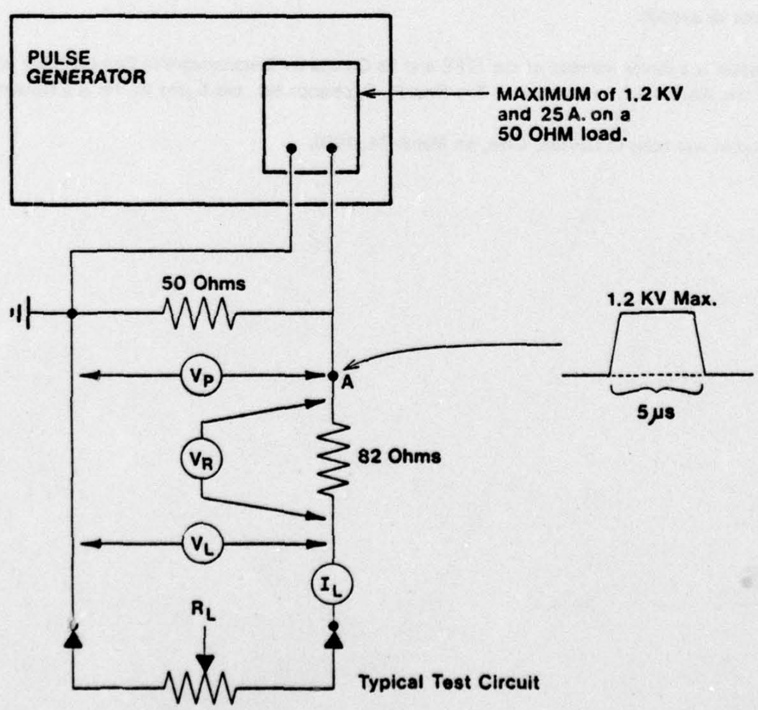

FIGURE 13 - TEST ARRANGEMENT FOR "HIGH" IMPEDANCE TEST CIRCUITS

QUESTIONS and ANSWERS

From D. Clifford

Q – The tests you describe deal entirely which the effects of current flow through the aircraft. The response measured on the wiring seems to be primarily high frequency in nature, bearing no correlation to either the di/dt or iR response described earlier by Brian. In view of the fact that flight measurements have shown the presence of H.F. response from nearby activity, where no current in flowing at all through the aircraft, I question whether the current flow is responsible for the response. How do you treat the circuit response which bear no apparent relationship to peak current or di/dt, in particular, how do you extrapolate it ?

A – Interior cable responses depend upon resonant effects of aircraft structure. Ground-based simulation can introduce distributed external circuits that resonate at low MHz frequencies when pulsed by capacitor discharge. Interior responses are spectrum dominated which accentuates higher frequency components. Linear analysis is applicable so extrapolation is valid in theory.

In-flight responses at high frequencies detected on interior test cables reflect impinging E and H field from nearby strikes which in-turn induce surface current (similar to nuclear EMP effects). Again, energy is coupled through apertures onto interior cables.

BIOGRAPHICAL NOTE

John C. CORBIN

Dr Corbin is a senior Electrical Engineer in the Atmospheric Electricity Hazards Group. Survivability/Vulnerability Branch, Vehicle Equipment Division, Air Force Flight Dynamics Laboratory (AFFDL). Prior to his AFFDL assignment, Dr. Corbin was assigned to the Office of the Assistant for Operations on the Air Force Wright Aeronautical Laboratories (AFWAL) Staff for one year. During the period from 1964-1975, he served as Deputy Director of the Plasma Physics Laboratory of the Aerospace Research Laboratories.

Dr Corbin received the Bachelor of Electrical Engineering degree from Cornell University in 1950 and the MS and PhD degrees in electrical engineering from Ohio State University in 1963 and 1974, respectively. He has authored articles and reports in the areas of skin effect, plasma wave instrumentation and control nonlinear optics, multiwave parametric processes, and assessment of atmospheric electricity hazards to aircraft.

Dr Corbin is a senior member of the IEEE and its Groups on Electromagnetic Compatibility and Engineering Management and is a member of the AAAS. He is a member of Tau Beta Pi, Eta Kappa Nu, and Sigma Xi. He is a registered Professional Engineer in Ohio.

Dr Corbin was born in Dayton, Ohio, on March 14, 1926.

DIRECT EFFECTS PROTECTION METHODS FOR THIN SKINS/COMPOSITES

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SUMMARY

The main increase in hazard from the direct effects of lightning currents resulting from the use of thin metal skins (i.e. less than 2mm thick) is mainly confined to the increased probability of arc root burn through. The technique of protection against this hazard by use of a thin ablating skin is discussed.

With the use of carbon fibre composites, (Graphite/Epoxy composites), there is an increased hazard due to heat damage and delamination at all interfaces and joints. This can be reduced by carefull design of the interface. At the arc attachment points greater energy input gives rise to more surface damage than with aluminium alloy, and there is also a risk of severe delamination, although the risk of complete burn through is appreciably reduced. This damage can be avoided by the use of a thin metal protecting ablation layer.

1. Introduction

The scope of this paper has been limited to the discussion of protection of thin metal skins and conducting composites only. Thus the protection of non conducting composites such as Radomes is not included. The most notable of the conducting composites is or course the Carbon Fibre Reinforced Plastic (CFRP), (Graphite/Epoxy Composite) and this therefore is the major consideration of this paper.

Work on assessing the attractiveness of CFRP to lightning has been carried out by several workers including McBrayer Skouby and Wienstock of McDonnel-Douglas USA(1) They conclude that "There is no measurable difference in attach point behaviour between metal and Gr/Ep composite".

Accepting the validity of this conclusion it would appear that a CFRP component would experience exactly the same probability and severity of lightning attachment as a similarly positioned metal component.

2. General Considerations

2.1 Thin Metal Skins

In general thin metal skins have no special hazards from direct effects compared with the more usual 2mm or more of aluminium alloy, other than the increased risk of burn through at the arc root, although of course the hazard from indirect effects may be considerably increased. Standard aircraft alloys can safely carry the full action integral of 2×10^6 A^2s with a total cross sectional area of only $14mm^2$ and it is unlikely that the current would be more restricted than that. The comments in this paper on thin metal skins will therefore be confined to the hazard of burn through.

2.2 CFRP Skins

The bulk resitivity of CFRP is in general between 2 and 3 orders of magnitude greater than aluminium alloy. There is therefore greater ohmic heating. The bulk material conductivity is anisotropic, and depends upon the general lay up of the bulk material. The conductivity along the fibre lay was measured by Scruggs and Gajda(2) as 2,000 mhos/m and more than two orders of magnitude less for currents perpendicular to the fibre. Work by Banks(3) indicated poor contact between adjacent fibres side by side across the width of the ply, but better contact between fibres above and below each other through the thickness of the ply or between plies.

High current densities can be achieved under pulse conditions, provided the total ohmic heating does not cause failure of the epoxy resin or within the fibres themselves. The failure mechanism is thermal, and the lowest temperature at which run away conditions can develope is about 65°C. Any section capable of carrying 2×10^6 A^2s without exceeding 65°C should therefore be capable of carrying the current from a severe lightning strike without damage. For most structures the dimensions required for structural purposes will be sufficient to give adequate current carrying capacity in the bulk material, and the main hazards exist at interfaces where current may be restricted to very small cross sectional areas or is forced to flow across interlaminate boundaries. This problem i~~s most likely to occur~~ at metal/CFRP or CFRP/CFRP interfaces.

3. Metal/CFRP Interfaces

Two basic problems arise at all metal/CFRP interfaces. Firstly there is the problem of transfering the current from the metal to the adjacent carbon fibre laminate, and secondly the problem of transfering current from that laminate to others comprising the total composite. There is little that can be done other than to design the interface so that the metal is in direct contact with as many of the various laminates as possible.

Various techniques may be applied to achieve this where large metal components are moulded in, the carbon fibres can be wound tightly around the metal in the process, thus ensuring good metal to carbon fibre contact. Where contact is made after moulding, the outer skin of epoxy should be removed from the CFRP by machining to expose the carbon fibres. One successful way of achieving this has been the use of countersunk titanium screws, the machined countersinking giving good contact to the different laminates of the composite. The traditional "wet" assembly techniques however can give rise to some problems, and the compatability of the metal to the CFRP also needs careful consideration.

Alternatively the initial contact between metal and CFRP can be made by first plating or vacuum depositing metal onto the machined CFRP face thus permitting the final joint to be made between two metal surfaces by traditional methods.

Glued joints need special attention. The glue itself is usually a good dielectric and is applied with a thickness adequate to present a fairly high level of insulation at the interface. Such glue layers also tend to have small voids which may ionise during lightning strike conditions. Current flowing through this ionized void causes rapid heating and expansion of the gas leading to delamination of the joint. Such delamination can take place at very low energy levels.

4. Arc Root Damage

Three main differences can be listed between the arc root conditions on Carbon Fibre panels, and metal panels. They are:-

1. The arc on metal panels is not contaminated with carbon plasma, whereas on CFRP it clearly will be. The burning voltage of the arc on CFRP will therefore be higher than on metal panels, and there will be a greater energy input from the arc root.

2. The increased resistivity of the CFRP gives greater ohmic heating in the immediate vicinity of the arc root, and in the nearby area.

3. CFRP has a much lower thermal conductivity than aluminium, and the much greater total energy imput due to 1 and 2 above cannot diffuse as easily as in aluminium, particularly where the heat has to cross a interlaminate boundary.

The energy input due to direct arc root energy is equal to $V_{arc}\int i\,dt$ and tends therefore to be coulomb-dependant. It will be up to an order of magnitude greater than the energy in an arc root to a metal plate.

The energy input due to ohmic heating is action-integral dependant. It will be between 2 and 3 orders of magnitude greater than the ohmic heating in a metal panel.

In the case of metal panels the significant damage viz burn through requires the current pulse to exceed both a minimum current and a minimum time dependant upon skin thickness before burn through can occur. For metal skin thickness of say 2 mm these conditions can only be achieved in the intermediate or continuing current phases.

In the case of CFRP panels delamination in the nearby areas can take place as a result of ohmic heating due to the action integrals of either the initial attachment currents, the intermediate currents, or the restrike currents, whilst directs arc root damage can arise due to the coulomb transfer of either the intermediate current or the continuing current phase.

The low thermal conductivity however reduces the depth to which heat damage extends and appreciably reduces the possibility of burn through.

The heat damage in the arc root takes the form of vapourization of the epoxy matrix with the subsequent tufting of the carbon fibres, and the damage in the nearby region (say 20 - 200mm from the arc root centre) may take the form of delamination of composite. For the most part the arc root heat damage would normally only penetrate 2 to 3 laminates deep except for very long duration discharges. The delamination area is similarly restricted and seldom exceeds 200mm.

Wind tunnel tests have shown that the tufting damage is not extended as a result of dynamic pressure from flight conditions, as the individual fibres have insufficient strength and break off under the wind forces. Delamination damage however is different and severe delamination damage can be extended by wind pressures. In general the surface damage is greater for CFRP than for metal, but the burn through hazard is less.

Mechanical tests have shown no detectable deterioration of mechanical properties of the test specimens outside of the area of visible damage.

5. Surface Protection

Work at Culham on surface protection of CFRP arose from earlier work on the protection of thin metallic skins against puncture. In this a fine metallic foil is applied to the outside of the thin skin with a thin insulating layer in between them, the insulating layer being at the same time the adhesive holding the outer foil to the inner skin. In principle lightning currents attach to the outer foil, the arc root moving over the sufaces as the foil ablates. The insulating layer protects the lower skin from the arc heat for a short period that the arc dwells over any given spot thus protecting the lower skin from arc root attachment. If the insulating adhesive layer is an organic material such as epoxy resin, the arc root will preferentially stay on the metal foil as a pure metal plasma arc has a lower burning voltage than a carbon contaminated arc. Experiments show that an outer foil as thin as 6μm with an insulating layer of similar thickness can give complete protection against the most severe test levels recommended.

This principle can be extended to CFRP and thin metal coating of as little as 6μm have been found to give excellent protection against arc root damage. The lower limit of 6μm can only be effectively used if the area covered is reasonably large (say $2m^2$ or more) and in many cases a layer of 15 - 20μm would be better. There seems little advantage in going much bigger, and in fact there may be some disadvantage, as the arc root dwells longer in one area with the thicker foils, and the risk of attachment to the CFRP is therefore increased.

The normal CFRP panel has a thin coating of epoxy resin on the outside which is normally adequate to produce the effect of the insulating/adhesive layer of the system. Direct application of metal coatings by plating or vacuum depositing may therefore be made, and this has also been completely successfull in experiments at Culham.

Almost any metal will do from the lightning point of view, and work has been done at Culham using Aluminium, Copper and Gold with equal success. The overwhelming bulk of the work however has been done with Aluminium. The protection can also be in the form of a wire mesh moulded into the surface. It has also been demonstrated that such protecting metal layers may be painted without loss of the protection offered.

Attempts at protection using metal powder loaded paints (Both aluminium and silver powders) have been complete failures.

Extensive work in this field has been carried out at ERA Leatherhead by White et al and other workers in Europe and the USA. Although this work shows some differences in detail from that at Culham, the general results are compatable.

6. Conclusions

(1) There is unlikely to be any serious direct effects hazard resulting from the use of thin metal skins, other than that of arc root burn through. Adequate protection against this hazard can be effected by the use of thin ablation layers of metal foil.

(2) The main hazard in zone 3 regions resulting from the use of CFRP is likely to occur at metal/CFRP or CFRP/CFRP interfaces. Careful design of these interfaces can appreciably reduce this hazard.

(3) There is an appreciable increase in the area of surface damage near the arc root with unprotected CFRP compared with aluminium alloy. It is not usually more than 2 or 3 laminates deep however and there is no invisible or latent damage outside of the visibly damaged area, and in many applications protection may not be required.

(4) Complete burn through of an CFRP is less likely than an aluminium panel of comparable thickness, even when unprotected, but serious delamination can take place under some circumstances.

(5) CFRP can be protected against arc root damage by a thin ablating layer of metal on the outside. This should be not less than 6μm thick, and may be painted.

References

(1) McBrayer Skouby and Wienstock - Lightning Attachment Charateristics for Metal/Composite materials. IEEE. EMC Conf. Seattle USA 1977.

(2) Scruggs and Gajda - Low Frequency Conductivity of Unidirectional Graphite Epoxy Composites. IEEE. EMC Conf. Seattle UAS 1977.

(3) Banks - CLSU Memo No.65.

Acknowledgments

The author wishes to acknowledge the support given by the Ministry of Defence for the work conducted at Culham Laboratory.

QUESTIONS and ANSWERS

1 – From D. Clifford

Q 1 – I'm surprised that you report that the contact resistance between carbon fibres between plies is less than between adjacent fibres in the same plies.

Q 2 – Has your ablation protection system actually been used in an operational aircraft ? How does the total weight compare to a metal skin thick enough to withstand the lightning DIRECTLY . . ?

A – The samples investigated at Culham were sheets of multiply carbon fibres. The contact resistance between the different plies was in fact lower than the contact resistance between adjacent fibres side by side through the width of the sheet, but it was a little higher than the contact resistance between adjacent fibres one above the other through the thickness of the ply. This is believed to be due to the direction and effect of the moulding pressures during manufacture. It should be noted however that this is only true of plies of different fibre directions. Where a number of adjacent plies were stacked together having the same direction of fibre lay, the contact resistance between adjacent plies was considerably increased.

In reply to your second question, the ablation system has been used on two aircraft. They are the Short Bros SD3-30 and CASA C212. In the first case the system took the form of a structural skin of ·040" (1 mm) with an ablating skin of .006" (.15 mm) giving a little over half the weight of a .080" (2 mm) skin, which is the minimum that could be considered adequate. In the case of the C212 an ablating layer of about the same thickness (viz. 0.15 mm) was applied to existing skins of 1.4 mm and 1.2 mm thickness. Again a thickness of 2 mm minimum would normally be required. In both cases the panels withstood a minimum of two discharges to the same point, each discharge in excess of 100 coulombs. This is far in excess of the withstand of 2 mm thick alloy.

2 – From B.L. Perry

Q – Why do the German tests produce large holes whereas the UK tests did not, with apparently similar tests on carbon fibre materials ?

A – The difference in results probably comes from a difference in test techniques. The Culham tests were conducted with jet diverting electrodes and relatively long arcs, whilst the German tests were conducted with normal metal test electrodes and very short arcs i.e. · 10 mm long with a 10 mm diameter electrode. This gives three effects which tend towards a pessimistic result. Firstly the test electrode jet puts very much more energy into the arc root on the test piece and secondly the arc is firmly rooted to one place and cannot move about. These two effects give much more heat damage to the test specimen than is the case with the longer arcs. Thirdly the rapid heating of the air in the confined space between test electrode and test specimen arising from the initial 200 kA pulse gives rise to a severe shock wave which is probably responsible for the impact fracture delamination seen on the reverse side of the specimens in the German tests. Both test methods of course give results which are useful for comparing different specimens with each other, but it should be noted that short arcs in general give pessimistic results.

Direct Effects, Protection Methods for Thin Skins/Composites

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Summary

As a result of the rapid development of fiber composite materials the German Ministary of Defense instructed in the course of the last years amongst other things Dornier, MBB and VFW the three aircraft system companies in Germany, as well as MTU, DFVLR and IABG jointly to investigate the applicability of fiber composite materials for aircraft engineering. Electrically nonconducting types of fibers, e.g. glass fibers for radar domes, antenna covers, helicopter rotor blades or wing tips are used for years.

A comparatively new procedure is the use of electrically conductive fibers like CFRP or BFC. Only the following examples shall be mentioned here

VFW 614:	glide-path spoiler (BFC construction)
VFW 614:	nose-gear door (CFRP/GFRP construction)
ALPHA JET:	air brake (CFRP construction)
MRCA:	taileron components (CFRP construction)

The destruction of fiber composite components due to lightning stroke is heavier than that of equivalent metal structures, therefore protection systems for this type of fiber composite materials have to be developed and used depending on the type of material or use.

A great advantage of fiber composite components as compared with metal components is the saving of weight. This advantage is partially neutralized due to the additional proportion of weight of lightning protection systems. In order to minimize this disadvantage this type of protection systems should be of minimum additional weight only.

Since electrically conductive fiber composite materials are used, the protection problem also has been of permanent concern.

According to the currently used test procedures a lightning protection system is referred to as sufficient if the protection system prevents the destruction of the fiber structure.

Damages of the lightning protection system and damages on the surface of the composite material are considered to be permissible.

Test Programme

Based upon an agreed test program between FR and FRG on lightning protection design techniques for advanced composite aircraft structures two protection systems of different technology have been tested:

a) the multilayer system

b) the surface mesh system

The following variations have been agreed between France and Germany:

	FRG							FR			
Sample	Carbon Fiber (HT) T 300/CIBA 914 8 layers (1mm thick, approximately) $0^o/90^o/0^o/90^o/90^o/0^o/90^o/0^o$ 40 x 40 cm							Carbon Fiber (HT) T 300/NARMCO 5208 8 layers (1mm thick, approximately) Orientation as for FRG			
Protection system	1	2	3	4	5	6	7	1	2	3	4
Protective painting	NLP	NLP	NLP	NLP	NLP	NLP	NLP	NLP	NLP	NLP	NLP
Upper layer	ALF	LP	ALF	FLA	-	ALF	-	-	-	-	-
Reactive layer	319	319	319	319	-	340	-	-	-	-	-
Lower layer	-	-	ALF	FLA	ALM	ALF	-	BZN	BZN	BZN	-
Weight in g/m^2								47 g/m^2	80 g/m^2	250 g/m^2	
Number of samples	15	15	15	15	15	15	15	3	3	3	3

NLP = nonconductive polyurethane primer

ALF = aluminium foil

FLA = aluminium, flame-sprayed

LP = conductive polyurethane primer

ALM = aluminium mesh

BZN = bronze mesh

319 = heat curing synthetic resin layer CIBA 319

340 = cold curing synthetic resin layer PR 340

Test Procedures

The composites were tested under defined and reproducable strokes using the laboratory test set-up at the Institute for Plasma Physics of the T.U. Hannover.

Defining uniform parameters the values listed below have been agreed:

	FR	FRG
Electrode diameter Ø	10 mm	10 mm
Distance between electrode and sample		
1. 200 kA/-	8 mm	10 mm
2. 50 kA/50 C/0,5 sec	8 mm	10 mm
3. 200 kA/50 C/0,5 sec	8 mm	10 mm
4. 200 kA/210 C/0,5 sec	8 mm	10 mm

As can be seen from the table three load test groups have been carried out in Germany.

First the tests using only one high intensity impulse of 200 kA without subsequent charge transfer.

Second the tests using 200 kA and a charge of 50 C for a time of 0,5 sec. Last the tests using 200 kA and a charge of 210 C for 0,5 sec have been carried out. In contrast to Germany the French side took the view that a stroke of 50 kA/50 C/0,5 sec would be more realistic with respect to natural lightning discharge than a stroke of 200 kA/50 CO,5 sec.

Above all the tests using 200 kA/210 C/0,5 sec have been carried through to come up to existing test conditions as close as possible and check for effectiveness the protection systems to be tested using a higher quantity of charge as well.

Method of Test

For the above described protection systems no. 1 to no. 7 some representative damage descriptions have been summarized. In this connection the protection procedures designated under 3, 4 and 5

3 aluminium foil

4 flame-sprayed aluminium

5 aluminium mesh

showed the best results.

The systems have been closely investigated within FRG to find further ways of optimization.

Furthermore there are plans to promote the exchange of experiences between FR and FRG and among other things to investigate in detail the bronze mesh protection system within FRG that is preferred by FR and to allow to FR a detailed investigation of the FRG optimized systems.

A correlation of the weights per unit area of the lightning protection systems to be tested including the primer weight shows an evident advantage of aluminium mesh.

Similar results have been received in France using bronze mesh.

Conclusions and Prospect

The tests carried through represent a beginning of further programmes. Besides from the optimization of lightning protection procedures which include among other things mechanical strenght tests of CFC samples after exposure to lightning strokes, considerations concerning corrosion resistance as well as further combinations of upper and reactive layers basic investigations in the form of research programmes will be started too on the physical destruction mechanisms apart from the purely engineering program above.

The methodical point of departure is considered to be the observation of the time history of the damage process in conjunction with the appropriate protection system. This allows a good observation of the important damages resulting from the first high-intensity current phase, i.e., the first 30 μs, by means of high-frequency cinematography.

This type of camera with rotating mirror system will be designed at the Plasma Institute of the TU Hannover, since there is no commercial camera available.

It is considered to use inert gas ion lasers with a performance of few watts to investigate the gas dynamic shock waves coming from the spark and to make the destruction visible, that is the evaporation and rupture of the CFC structure or the protective system, respectively.

For this an extraneous light source should be used, since only the current-carrying are channel will light up intensively.

As a possible conclusion one expects among other things findings on the elastic plasticity of protection system, physical limits of ablation protective layers and clues as to optimal mesh systems.

Technically and economically the use of new fiber reinforced material for current and future aviation and space projects will only be practical of one answers as much as possible or one would almost say all of the engineering problems related to this new technology.

References

Entwicklung von Blitzschutzsystemen für Bauteile an faserverstärkten Werkstoffen ZTL 1977-78 FAG-2.

Dr. M. Flemming, Entwicklung und Anwendungsmöglichkeiten von Bauweisen aus faserverstärkten Werkstoffen, Dornier GmbH.

Appendices

A-1 to A-6

Report of flash protection systems

B-1 to B-3

Description of damage

C-1

Correlation of weights per unit area of different flash protection systems including primer weights (1977).

Test Number 92 System Number II

Notes: 176 KA, 210 C, 0,53 sec.

Sample Face:

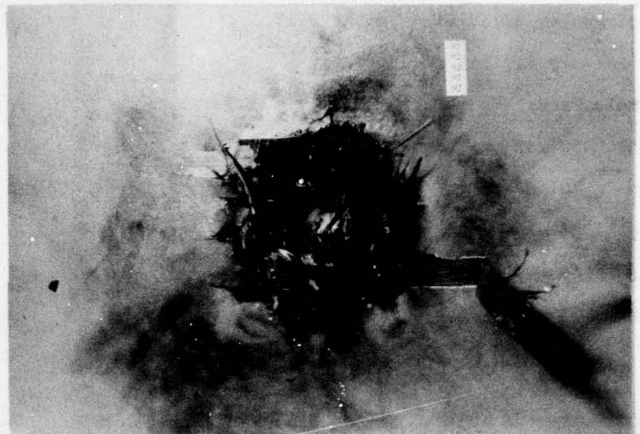
NLP

LP 150/80 mm cracks

319A

US-

CFK 35 mmØ hole



Test Number 92 System Number II

Notes: 176 KA, 210 C, 0,53 sec.

Back of Sample:

90 mmØ cracks

35 mmØ hole

Test Number 43 System Number III

Notes: 220 KA, 239 C, 0,54 sec.

Sample Face:

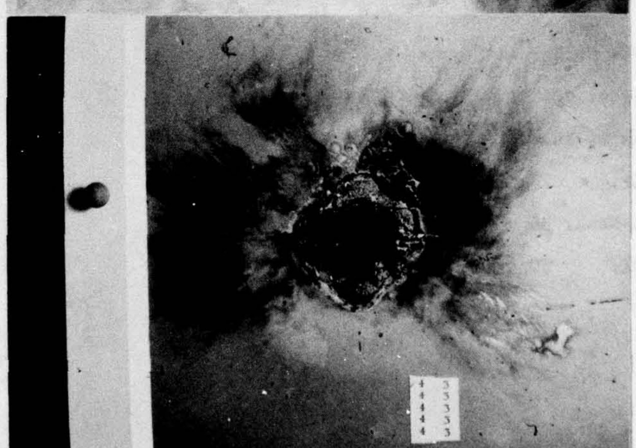
SS: NLP 55 mmØ, peeled off

OS: ALF 50 55 mmØ, molten

RS: 319A 50/40 mm burned

US: ALF 25 50/40 mm molten

CFK 10 mmØ, hole



Test Number 43 System Number III

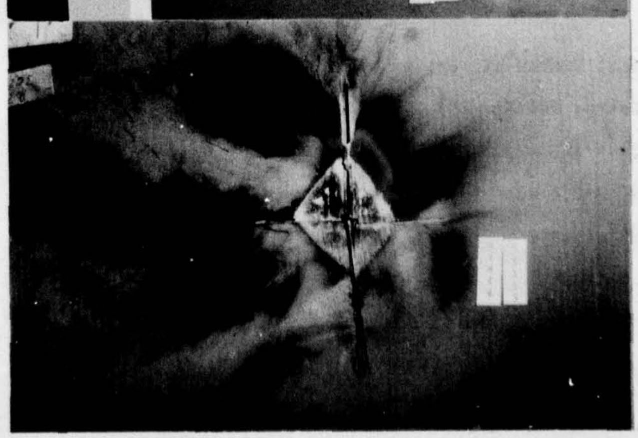
Notes: 220 KA, 239 C, 0,54 sec.

Back of Sample:

50 mmØ burned area

120/80 mm $0^{o}/90^{o}$ cracks

10 mmØ hole



Test Number 44 System Number B IV

Notes: 213 KA, 239 C, 0,54 sec.

Sample Face:

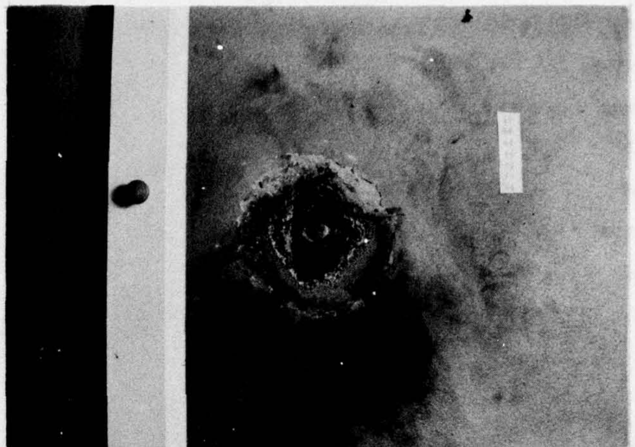
NLLP 70 mmØ peeled off

FLA 70 mmØ peeled off

319A 40 mmØ burned

FLA 40 mmØ burned

CFK 15 mm $90^{\circ}/0^{\circ}/90^{\circ}/0^{\circ}$ cracks



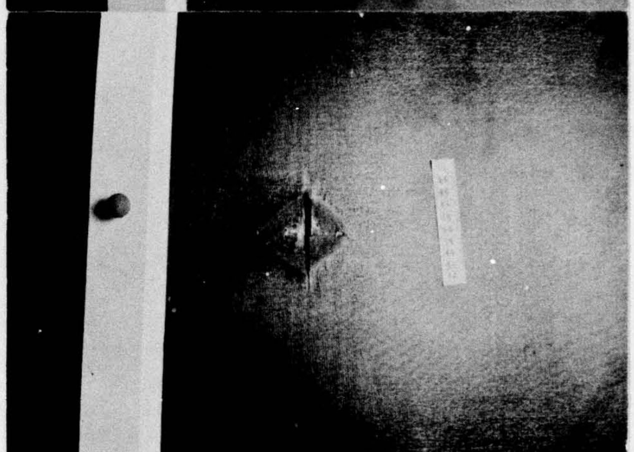
Test Number 44 System Number B IV

Notes: 213 KA, 239 C, 0,54 sec.

Back of Sample:

45 mmØ roughened surface

50 mm burned crack



Test Number 41 System Number V

Notes: 206 KA, 233 C, 0,54 sec.

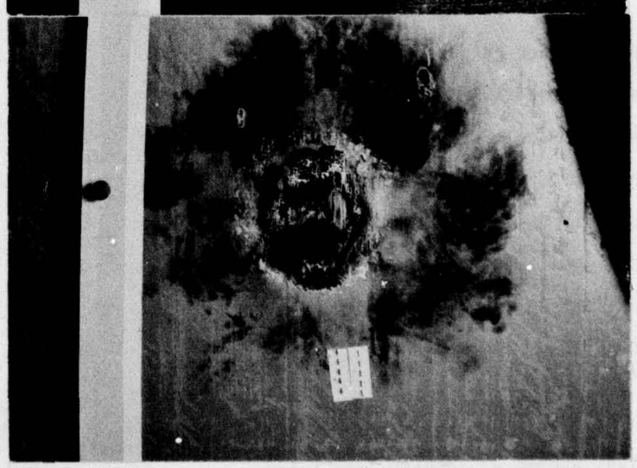
Sample Face:

NLP: 70mmØ peeled off

Al-Mesh: 50/65 mm molten

CFK: 50/65 mm burned

15 mmØ $90^{\circ}/0^{\circ}/90^{\circ}/0^{\circ}$ cracks



Test Number 41 System Number V

Notes: 206 KA, 233 C, 0,54 sec.

Back of Sample:

60 mmØ roughened surface

90° cracks

no picture

Test Number 57.1   57.2

System Number C VI

Notes: 213 KA, 46 C, 0,5 sec.

Sample Face:

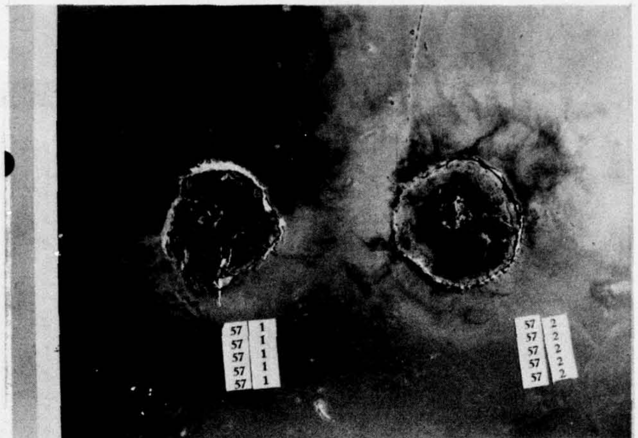
NLP 55mmØ peeled off

ALF 50 50 mmØ molten

PR 340 15 mmØ burned

ALF 25 10 mm molten

OFK   10 mmØ   0°/90° cracks



Test Number 57.1 57.2

System Number C VI

Notes: 213 KA, 46 C, 0,5 sec.

Back of Sample:

25/50 mm 0°/90° cracks

Test Number 36

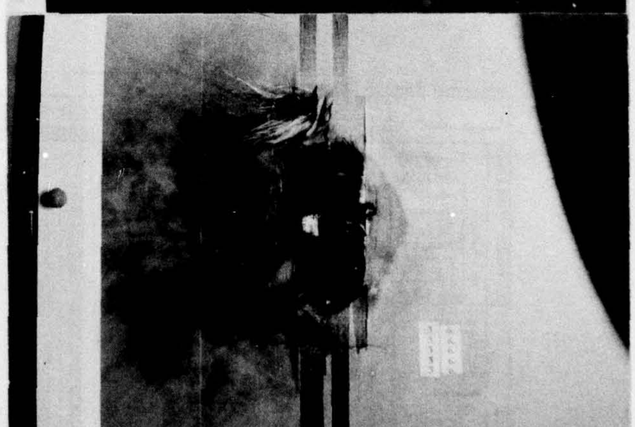
System Number VII

Notes: 169 KA, 46 C, 0,5 sec.

Sample Face:

NLP 80 mmØ burned

CFK 10/60 mm hole burned



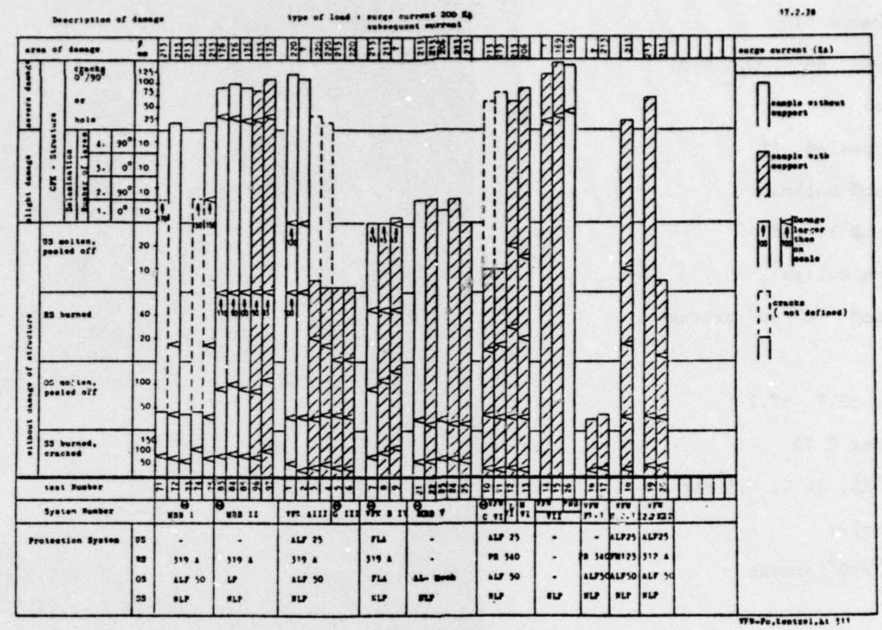
Test Number 36

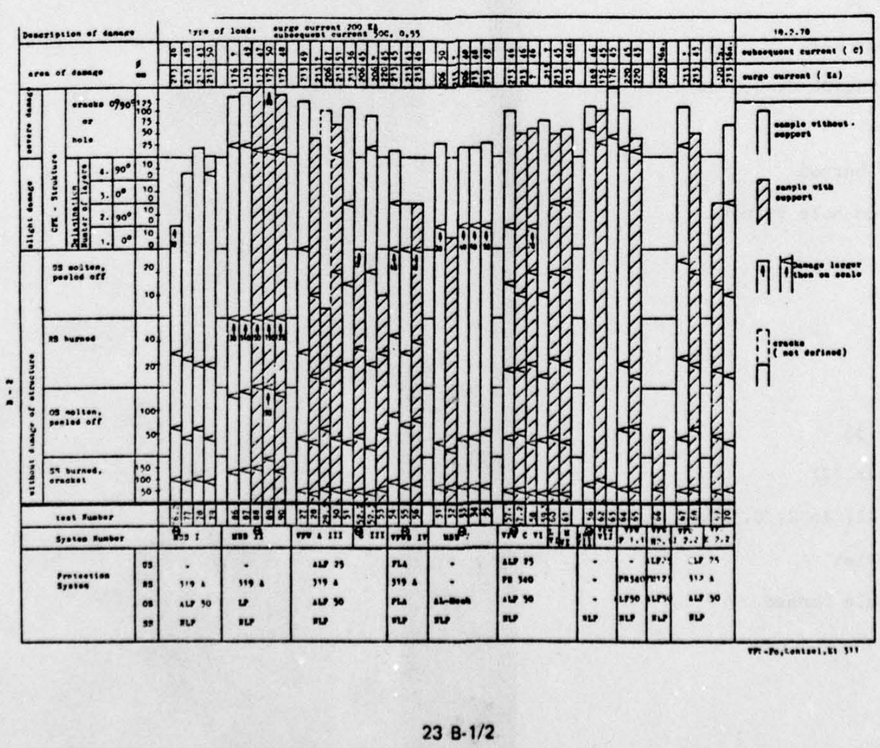
System Number VII

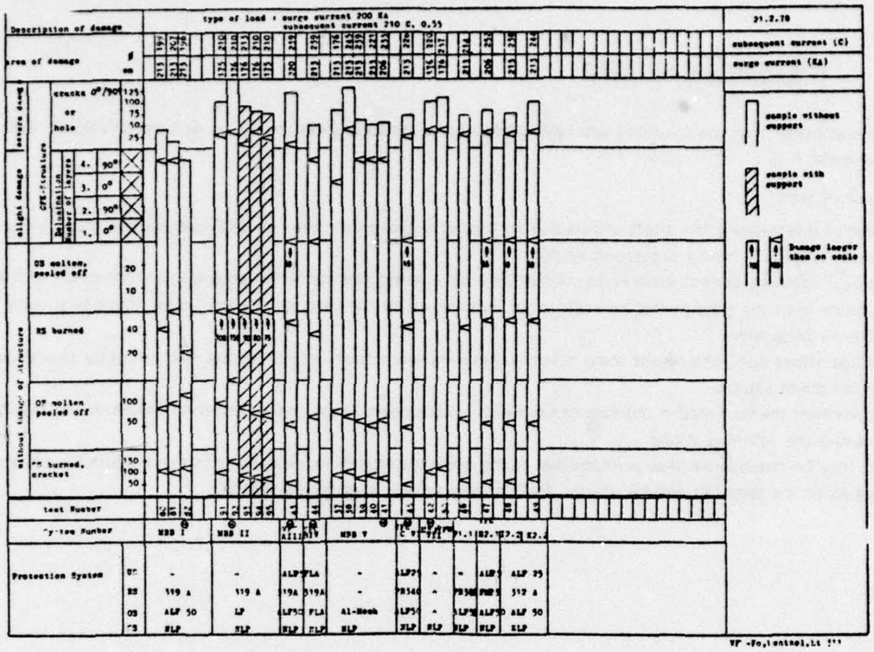
Notes: 169 KA, 46 C, 0,5 sec.

Back of Sample:

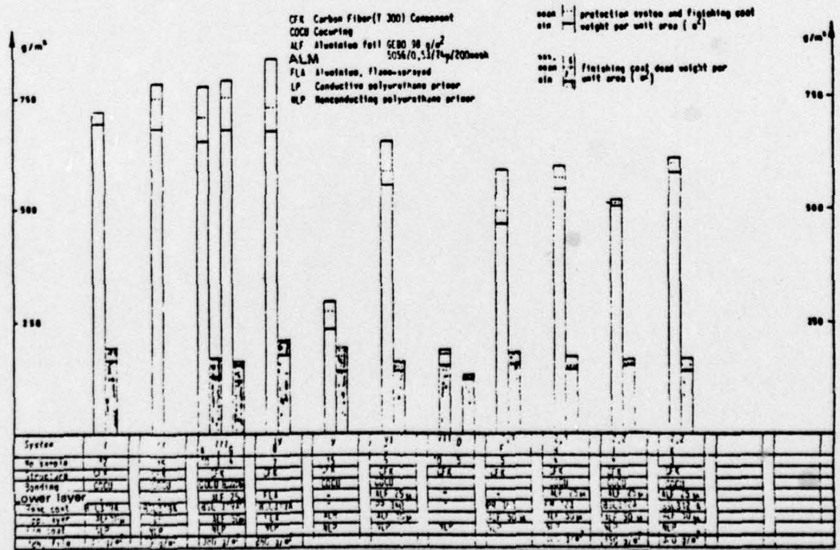
70/110 mm hole burned


Description of damage
type of load : surge current 200 kA
subsequent current
17.2.78
area of damage
surge current (kA)
severe damage
slight damage
without damage of structure
CFK - Structure
Delamination
Number of layers
cracks 0°/90°
or
hole
US molten, peeled off
RS burned
OS molten, peeled off
SS burned, cracked
test Number
System Number
Protection System
sample without support
sample with support
damage larger than on scale
cracks (not defined)
B - 1


Description of damage
type of load: surge current 200 kA
subsequent current
subsequent current (C)
surge current (kA)
area of damage
severe damage
slight damage
without damage of structure
CFK - Structure
cracks 0°/90°
or
hole
US molten, peeled off
RS burned
OS molten, peeled off
SS burned, cracked
test Number
System Number
Protection System
sample without support
sample with support
damage larger than on scale
cracks (not defined)
B - 2



B - 3



C - 1

MBB Correlation of weights per unit area of different flash protection systems including primer weights

QUESTIONS and ANSWERS

Q – Do you consider that the solutions you have submitted to tests are good enough to be used in aircraft, even in the most lightning critical areas ?

A – Purpose of tests

The purpose of investigating the direct effects due to lightning stroke by means of CKF-samples was initially mainly the selection of a most suitable lightning protection system.
The selected critical lightning current wave form constitutes a very severe test condition accepting a destruction of the specimen.
To draw conclusions from the tests carried through so far with regard to the usability for individual lightning striking zones in the aircraft has not been considered.
From these considerations only, the results show that the lightning protection systems investigated are quite able to provide satisfactory protection for direct effects.
We have not considered the fact that in the case of sandwich constructions, type and structure of the sandwich may be decisive for the behaviour during the lightning stroke.
Summarizing, it may be noted again that is impossible at the present status of investigations to make a satisfactory statement, whether the available results of the tests can also be utilized for the most critical areas of the aircraft.

PROTECTION METHODS FOR HARDWARE

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SUMMARY

Present understanding of aircraft lightning effects and the means that are available to protect against them has advanced to the point where it is possible to design an aircraft to be safe from hazardous lightning effects. Lightning protection is, of course, most effective, easiest to incorporate and least expensive when designed into an aircraft while it is still on the drawing board. Lightning protection can often be retrofitted onto existing aircraft, but the results are rarely as thorough, and the process is usually more costly.

For most aircraft, adequate lightning protection influences the design of each of its major systems and structures. The designers of each system and structural element must be made aware of potential lightning problems and the resources that are available to help solve them. Many problems that have arisen in the past were due to lack of designer awareness rather than to inadequacies in available protection technology. Unfortunately, this technology is documented in numerous technical reports and references whose existence is unknown to many designers. Since space does not permit description of hardware protection methods in this paper, sets of checklists are provided for typical aircraft systems together with references to sources of further information. Hardware addressed in this paper includes externally mounted components, non-metallic structures, fuel system hardware, and control surfaces. Examples of some of the more common problems are also given. Protection methods for thin skins and composites are addressed in Papers Nos. 22 and 23, and protection for electronics is discussed in Paper No. 25.

Basic Steps in the Design of Lightning Protection

As discussed in Paper No. 7, the basic steps in design of lightning protection are as follows:

1. Determine the Lightning Strike Zones

   Determine the aircraft surfaces, or zones, where lightning strike attachment to the aircraft is probable, and the portions of the airframe through which lightning currents must flow between these attachment points.

2. Establish the Lightning Environment

   Establish the component(s) of the total lightning flash current to be expected in each lightning strike zone. These are the currents that must be protected against.

3. Identify Vulnerable Systems or Components

   Identify systems and components that might be vulnerable to interference or damage from either the direct effects (physical damage) or indirect effects (electromagnetic coupling) produced by lightning.

4. Establish Protection Criteria

   Determine the systems and/or components that need to be protected, and those that need not be protected, based upon importance to safety-of-flight, mission reliability or maintenance factors. Establish lightning protection pass-fail criteria for those items to be protected.

5. Design Lightning Protection

   Design lightning protection measures for each of the systems and/or components in need of protection.

6. Verify Protection Adequacy by Test

   Verify the adequacy of the protection designs by laboratory qualification tests simulating the lightning environments established in step 2 using the pass-fail criteria of step 4.

The lightning environment to which each object should be designed depends, of course, on the particular lightning zone in which it is located. Environments expected in each zone are presented in Paper No. 7. The information in this paper deals primarily with steps 3, 4 and 5. Checklists of the lightning protection problems that may be encountered are presented in Tables I through IV, together with references to sources of further information concerning each problem. A few illustrations are presented as examples.

TABLE I - External Hardware

External hardware includes air data probes, antennas, radomes, navigation lights, windshields, canopies and other objects mounted on the external surface of an aircraft. Since many of these objects are in lightning strike zones, they must be designed to safely conduct lightning currents and to prevent surges from being coupled into associated electrical wiring, if present. Particular areas of concern are are as follows:

Air Data Probes (Including pitot probes, angle-of-attack probes, alpha probes, etc.)	References
1. Are these probes located in lightning strike zones? If so,	
a. Can the probe mounting bolts or other fastening means conduct lightning currents into the airframe without physical damage?	1, 3, 4
b. Are probe heater and signal circuits protected from IR voltages that develop as lightning current passes through fasteners?	2, 5
c. Are probe heater and signal circuits shielded from lightning magnetic fields to protect against induced voltage surges?	
d. If the answer to (c) is no,	
1. Are surge suppressor(s) provided to limit induced voltages to safe levels?	2, 5
2. Is series impedance or other means provided to limit surge currents to safe levels?	5
e. If the answer to (c) is yes,	
1. Is the magnetic shield (usually a braid or rigid conduit) adequate to conduct lightning currents?	6
2. Is the resistance of the shield low enough to prevent excessive temperature rise and/or IR voltages which may appear in internal circuits?	7

Example

A situation that has resulted in the loss of several military aircraft and extensive damage to electronics within many others is illustrated in Figure 1.

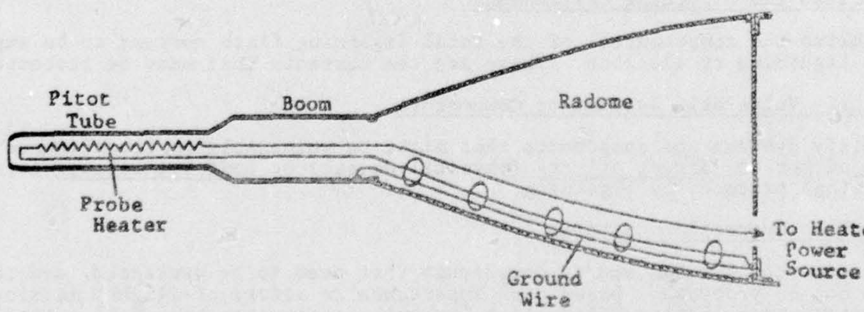

Figure 1 - Unprotected Probe Heater Circuit.

In Figure 1, the magnetic field produced by lightning current flowing in the ground wire induces a voltage surge in the heater power circuit. A surge also appears at the ungrounded probe heater, causing breakdown of the heater insulation and failure of the heater element. On occasion the ground wire is inadequate to conduct severe lightning currents and it explodes, severely damaging the radome.

An example of the voltage that may be induced in an unprotected heater power circuit such as that of Figure 1 is shown in the laboratory test of Figure 2. In this test, a mild stroke of only 5 kiloamperes induced 1500 volts in the circuit.

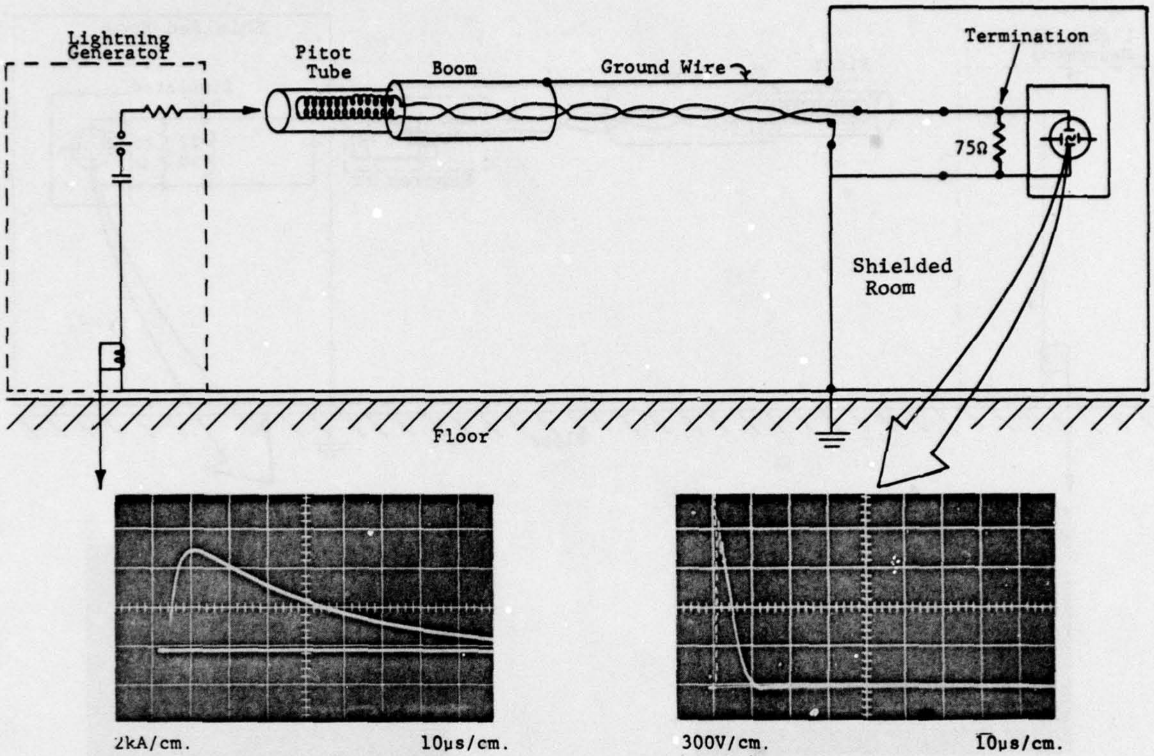

Figure 2 - Laboratory Test Showing Induced Voltage in an Unprotected Probe Heater Circuit.

There are several ways to protect against this effect. If a metal conduit can be tolerated, this conduit can serve as the lightning current conductor and the heater wires may be run within it where the magnetic field is (theoretically) zero. The IR voltage developed as lightning current flows in the conduit will appear in the heater circuit if one side of it is grounded at the probe, but this voltage can be minimized by selecting a conduit of sufficiently low resistance.

If a metal conduit is not feasible (i.e. if its presence interferes with adequate operation) a surge suppressor can be used. This suppressor must then perform these functions:

- Suppress the induced voltage to a tolerable level
- Reduce lightning currents in the heater power circuits to tolerable levels

Accomplishment of these functions usually requires a suppressor with several components; including series impedance to limit the lightning current (and force it to flow in an alternate ground wire); and a shunt suppressor to clamp the voltage surge. A laboratory test of such a suppressor is shown in Figure 3.

In the test of Figure 3, the AC power source was simulated by a 20 μH inductor and a 75Ω resistor.

	References
Antennas (Including long-wire, blade and dish types)	
1. Are these antennas located in Lightning Strike Zones? If so,	
a. Can the antenna mounting bolts or other fastening means conduct lightning currents without damage?	1, 3, 4
b. Are the antenna cables protected from lightning currents and voltages that may enter the antenna circuit and damage communication or navigation equipment?	12
c. Are dielectric antenna covers or fairings adequately protected against puncture and physical damage? If not can such damage be tolerated?	8, 9

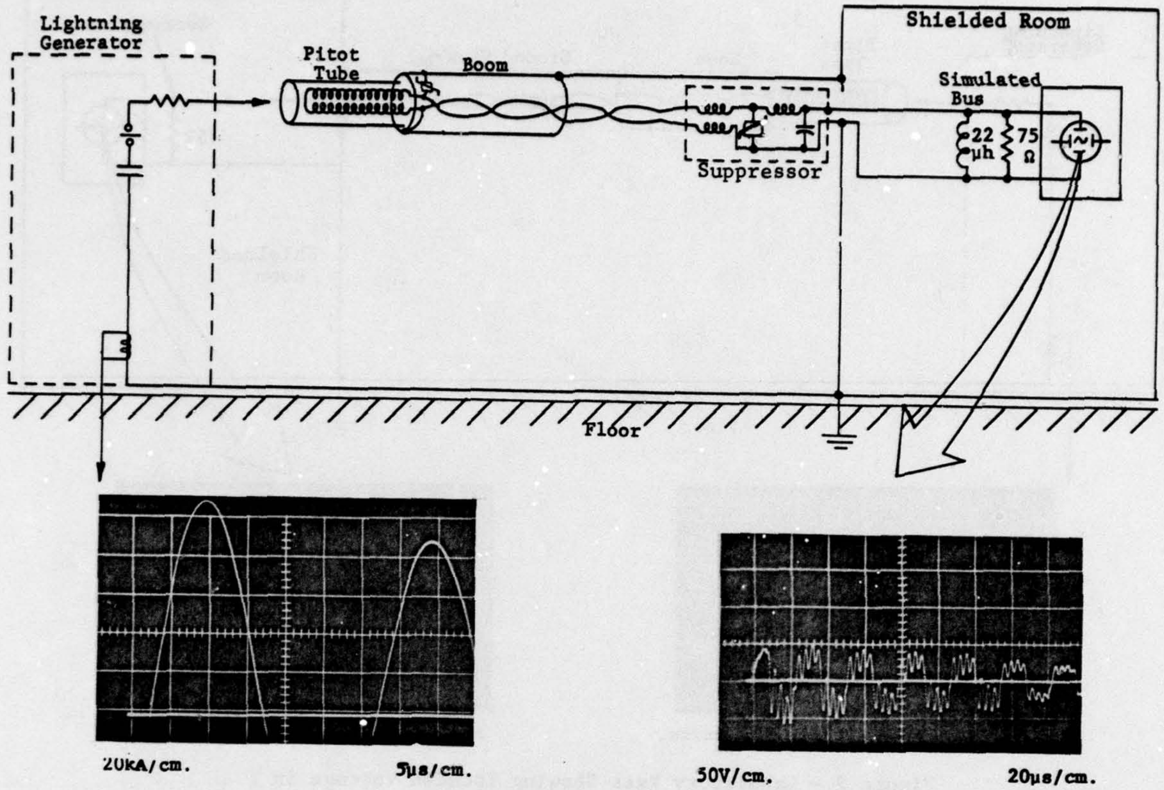

Figure 3 - Laboratory Test Showing Induced Voltage in a Protected Probe Heater Circuit.

<u>Radomes and Other Dielectric Covers</u>
(Including radomes, ECM pads, antenna fairings, wing and empennage tips and other dielectric covers)

	References
1. Are radomes located in lightning strike zones? If so,	
a. Is the radome protected against puncture from a lightning strike?	8, 9, 10, 11
b. If diverter strips are utilized to protect against puncture, can these strips, themselves, tolerate the required lightning strike currents?	8, 9, 10, 11
c. Are the diverters compatible with the operation of the radar antenna? Can they tolerate other environments such as rain and hail?	8, 10, 11

<u>Navigation Lights</u>
(Including lights mounted on vertical fins, wing tips, and fuselage)

1. Are the lights located in lightning strike zones? If so,	
a. If the lamp is mounted on a non-metallic structure is its housing grounded to metallic structure by conductors large enough to safely conduct the lightning currents?	1, 7
b. Can the lamp globes resist breakage from the lightning blast effects? If breakage may occur, are the bulb and its power conductors adequately protected against direct contact with the lightning currents?	13
c. Are the lamp power conductors adequately shielded from lightning magnetic fields?	2, 14

<u>Example</u>

A situation that has resulted in severe damage to wing tip structures as well as the aircraft's electric power system is pictured in Figure 4.

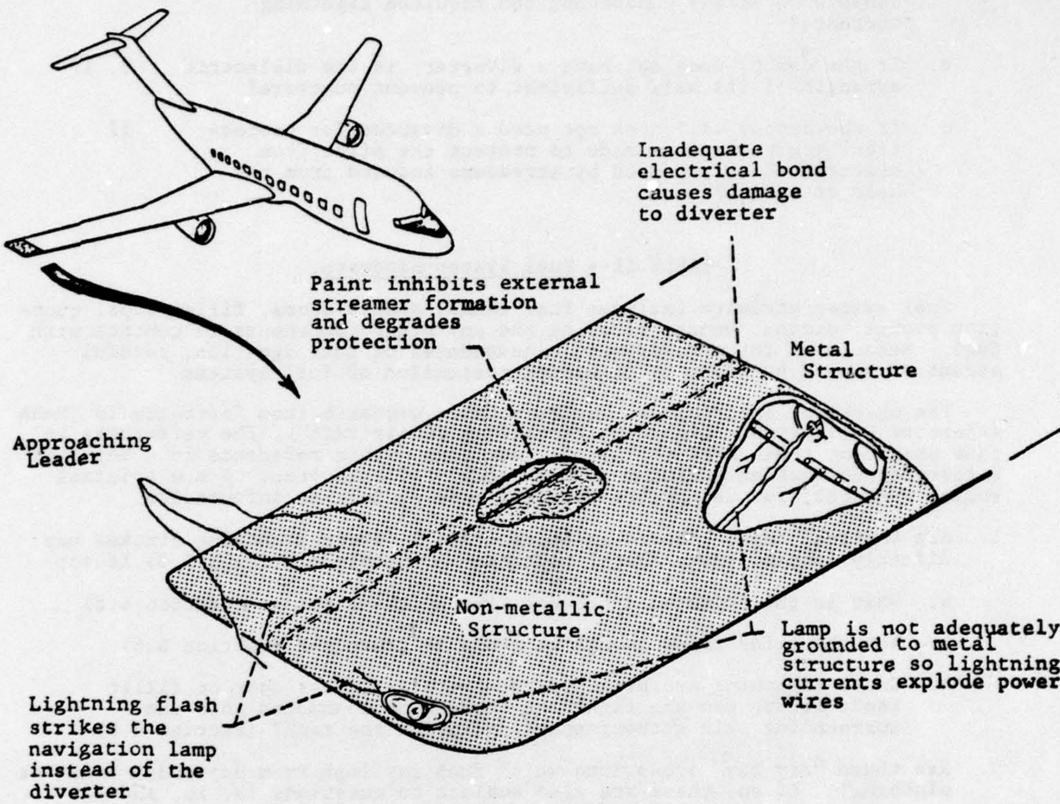

Figure 4 - Navigation Lamp Mounted on a Non-metallic Wing Tip.

The solution to this problem is to:

- Remove the paint from the diverters
- Provide a conductor large enough to safely conduct lightning currents from the lamp to metallic structure. The diverter strip may be used for this purpose if it provides the path of least inductance.
- Provide magnetic shielding and/or surge suppression for the lamp power circuit, using the same techniques as previously described for protection of probe heater circuits.

<u>Windshields</u>
(Including transport aircraft windshields with electric heaters) — **References**

Most windshields and canopies are located in either a direct or swept lightning strike zone.

a. Can the entire windshield, or its outer laminate, be punctured by the high voltages associated with a lightning strike? If so, — 15, 18

   1. Has adequate protection been provided against shattering glass, decompression or other resulting hazards to the crew?

   2. Can protection be provided against puncture?

b. If the windshield is electrically heated, have the heater power and control circuits been provided with surge suppressors to keep induced voltages and current surges from entering the aircraft's electric power distribution system? (note: Shielding is often impractical due to the necessary exposure of the heater element). — 15, 18

c. If a windshield center post is present, are electric wires that may be routed behind it adequately shielded against the concentrated magnetic fields that may be present when the center post is struck by lightning? — 2

Canopies (Including bubble-type canopies for fighter aircraft)	References
a. Does the canopy have an external diverter bar to protect against puncture of the canopy? Is this diverter bar capable of safely conducting the required lightning currents?	7, 17
b. If the canopy does not have a diverter, is the dielectric strength of its wall sufficient to prevent puncture?	9, 17
c. If the canopy wall does not need a diverter for protection, are provisions made to protect the pilot from electric shock produced by streamers induced from his head or helmet?	17

TABLE II - Fuel System Hardware

Fuel system hardware includes fuel tanks, access doors, filler caps, quantity probes, drains, vents, plumbing and any other components in contact with fuel. Because of the catastrophic consequences of fuel ignition, careful attention should be given to lightning protection of fuel systems.

The checklist that follows has been taken verbatim from Reference 16 (NASA Reference Publication "Lightning Protection of Aircraft"). The references in this checklist thus refer to specific sections within reference 16. The NASA reference publication includes a large number of references to the original source material, for use by the designer seeking further information.

1. Are the fuel tanks located in Zones 1 and 2, where lightning strokes may directly contact their skins? (Chapter 5, Sections 5.2 and 5.3) If so,
   a. What is the expected arc dwell time on these skins? (Section 6.6)
   b. Are the skins thick enough to avoid meltthrough? (Section 6.6)
   c. Can a lightning arc attach directly to an access door or filler cap?; if so, can the lightning current be conducted into the surrounding skin without sparking inside the tank? (Section 6.8)
2. Are there "dry bay" areas into which fuel may leak from adjoining tanks or plumbing? If so, these are also subject to questions 1a, 1b, and 1c.
3. Are fuel vent outlets or jettison pipes located in direct strike zones (Zones 1A or 1B) or swept-stroke zones (Zones 2A or 2B) where the arc may attach or sweep close to the vent outlet? (Chapter 5, Sections 5.2 and 5.3; Sections 6.4 and 6.5)
   a. If so, is an effective flame arrester or surge tank protection (STP) system used?; has its effectiveness been verified by test? (Section 6.4)
   b. Is the response time of the STP system shorter than the possible flame propagation time from the STP sensor to the extinguisher? (Section 6.4)
   c. Is the STP system protected from false trips resulting from lightning-induced voltages in its electrical wiring or light from nearby flashes?
4. If nonmetallic skins are used, is an adequate diverter or conductive coating system that will prevent skin puncture or internal streamering provided? (Section 6.7)
5. Is the fuel tank structure capable of *conducting* Zone 3 lightning currents even if the tank itself is not located in a direct or swept-stroke zone? Has this been demonstrated by a test in which simulated lightning currents are conducted through a complete tank structure? (Section 6.8)
6. If a simulated lightning test of the complete tank assembly is not feasible, have all of its individual joints, seams, access doors, filler caps, drains, vents, plumbing, and electrical systems been tested for their ability to conduct simulated lightning currents and found to be free of sparking? (Sections 6.5 and 6.8)
7. Are electric circuits entering the fuel tanks protected against high induced voltages? (Section 6.8) Have they been routed away from other wiring, such as navigation lamp circuits, which may be susceptible?

8. Are clearances between exposed electrical parts and each other or between any of them and the airframe sufficient to withstand the induced voltages that may occur without sparking? (Section 6.8)

9. Have the electrical circuits inside nonmetallic tanks been adequately shielded so that excessive induced voltages will not occur? (Section 6.8)

10. Have applicable government airworthiness certification regulations, military standards, or other specifications pertaining to lightning protection been adhered to? (Section 5.4)

TABLE III - Flight Controls

Flight control hardware includes the control surfaces themselves, as well as associated hinges, bearings, cables and actuators. Also included are electronic controls, but protection of electronics is dealt with in Paper No. 25.

References

1. Are control surfaces located in lightning strike zones?

   a. Are control surfaces themselves adequately protected against structural damage due to direct or swept strikes?

   b. Are the control surface hinges or bearings capable of conducting the required lightning currents without damage or freeze-up? — 19, 20

   c. Is it possible for lightning currents to enter control surface actuators and cables? If so, are the actuators and cables able to conduct these currents without damage or freeze-up?

TABLE IV - Propellers

Propellers include those on fixed-wing aircraft as well as helicopters. Due to their size, most propellers are within a lightning strike zone.

1. If the propeller blades are nonmetallic, or if they include nonmetallic components,

   a. Has an adequate path been provided for lightning currents to be conducted across or through the blade and into the airframe without damage to the blade, control linkages or bearings? — 21, 22

   b. Has adequate protection been provided for the blade heater circuit (if present) and have its electric power source and controls been provided with surge suppression to protect against induced voltages?

2. If the propeller blades are metallic, the possibility of lightning damage is greatly reduced but not eliminated, especially if adhesive bonding is utilized for fastening metallic sections together. Thus, the ability of metal blades (and associated bearings and control linkages) to conduct lightning currents must also be verified, and consideration must be given to protection of blade heater and prop pitch control circuits, etc.

Protection of Hardware within a Nonmetallic Aircraft

Fiberglass and graphite reinforced plastics (composites) are beginning to replace aluminum in aircraft skin and structural applications. Protection of these composites is the subject of papers Nos. 22 and 23, but even if the composites themselves can be designed to withstand severe lightning currents, protection of metallic hardware within may become a more difficult task. The reason is that the lightning current will deliver much more energy to a composite aircraft than to one built of aluminum.

For example, a measure of lightning intensity is its ability to deliver energy, or action integral. The action integral for a severe lightning stroke (which an airframe should be designed to tolerate) is 2 million ampere^2-seconds. The amount of energy deposited from the stroke is obtained by multiplying the action integral by the resistance of the conductor the current is flowing through - in this case the aircraft skins and structure.

The graphite composites now appearing have much more resistance than aluminum. For example, the electrical resistivity of aluminum is only 2.8 microhm-centimeters, whereas the resistance of graphite is over 500 times greater at about 1500 microhm centimeters. Since the amount of electrical energy deposited in an airframe is proportional to its resistance, an airframe built of graphite will receive nearly 500 times as much energy from a lightning strike as one of aluminum. An estimate of the nose-to-tail or wing tip-to-wing tip resistance of a fighter aircraft built of graphite would be about 50 milliohms.

The aircraft would then receive:

$$\begin{aligned} \text{Energy} &= (\text{Action Integral})(\text{Resistance}) \\ &= (2\times10^{6}\,\text{amp}^{2}\text{-sec})(50\times10^{-3}\ \text{ohms}) \\ &= 100{,}000\ \text{watt-seconds} \end{aligned}$$

of energy. This sounds a bit worse than it really is; in practical terms it is equivalent to the energy released by a one-kilowatt electric heater in a minute and a half. If it can be distributed evenly enough along the structure, this amount of energy should not be harmful to the composite material. Making sure of this is one of the designers tasks. Making sure that adhesive bonds do not degrade as currents flow between structural elements is another.

These are not the only tasks, however, because even when the structure is safe, voltage differences will arise as the lightning current flows through its resistance. If the nose-to-tail resistance were indeed 50 milliohms, the voltage difference along the fuselage would be:

$$\begin{aligned} V &= i_L R \\ &= (200\times10^{3}\ \text{amps})(50\times10^{-3}\ \text{ohms}) \\ &= 10{,}000\ \text{volts} \end{aligned}$$

assuming 200,000 amperes of lightning current. Similar voltages will arise when lightning currents flow through the wings or empennage. By itself, this voltage is not as harmful as it may seem because it is distributed along the length of the structure, but it will drive currents into interior conductors such as control cables, electrical wiring, hydraulic lines, and nearly any other parts that are made of conducting materials and in contact with the composite structure. This is where some new problems lie, since these parts are not normally designed to conduct electric currents. If only one or two amperes (of the up to 200,000) go astray and cause an electrical spark within the fuel tank, a catastrophe could result. Thus the lightning protection design must deal not only with the structure, but with nearly everything within it as well. This is why designers of nearly all of the systems within a composite aircraft must be made aware of the potential lightning effects upon their systems.

For example, hardware items within an integral fuel tank covered with composite skins must now be designed to safely conduct lightning currents, as shown in Figure 5.

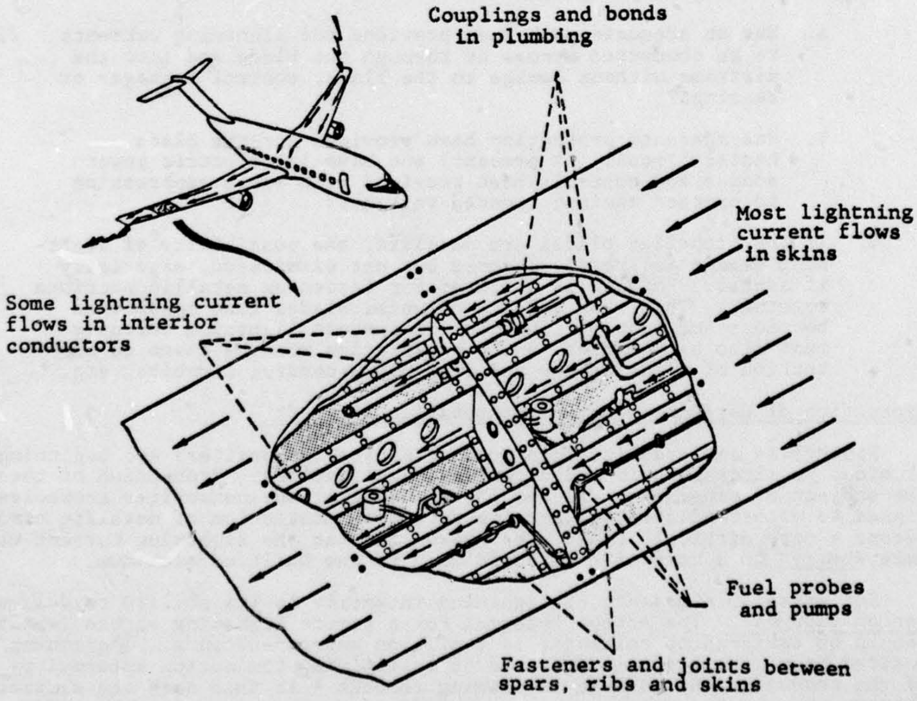

Figure 5 - Hardware within a Composite Fuel Tank Must Be Designed to Tolerate Lightning Currents without Sparking.

Conclusion

The checklists and examples presented in this paper are intended to alert designers to potential problem areas associated with aircraft hardware. Designers should use them as guidelines to help identify actual components in need of attention in particular aircraft.

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QUESTIONS and ANSWERS

1 – From S.D. Schneider

Q – Your Table III on flight controls indicated that bearings should be checked for freeze-up. I know of no incidents where bearings have welded to lightning. Do you ?

A – Yes. In the past, bearing freeze-ups have been reported on control surface bearings of small aircraft and gliders, although details are scanty. I know of no cases of bearing freeze-up on transport-category aircraft, probably due to the comparatively large size and location of these bearings on large aircraft. In cases where small bearings might be exposed to high current densities it will be prudent to evaluate freeze-up possibilities by simulated lightning tests.

2 – From G. Odam

Q – Freezing of bearings discussion : Do you agree that bearing freeze-ups are more important with respect to rotary wing head control bearings ?

A – Yes.

3 – From P.F. Little

Q – (comment) : Current flow in bearings can initiate damage which leads to rapid wear in subsequent use. The life of constantly-working bearings is considerably reduced by this effect.

A – Concur.

Also, the following three comments were received with respect to electrostatics and fuel systems.

1) From M.C. Jarvis

Q – (comment) : I am amazed that instrumentation is being developed to detect the smallest of sparks in fuel tanks when civil airlines are permitted to refuel with passengers aboard when the spark activity is many, many more orders greater.

A – (reply) : The degree of spark activity within a fuel tank during refuelling may indeed be much greater than that which occurs during a lightning strike (and still not cause an explosion) but the energy released by a spark of latter origin may be significantly greater, and sufficient to ignite a hydrocarbon vapor. This is the reason for the interest in detection of internal sparks during simulated lightning testing of aircraft fuel tanks.

2) From M.C. Jarvis

Q – (comment) : Concerning in-flight refuelling. Streamers on tanker hose and causing ignition of fuel.

A – (reply) : I know of no case where a spark or streamer at the end of a tanker hose (refuelling boom) has caused ignition, although this possibility should always be given careful consideration in the design of in-flight refuelling systems.

3) From J. Nanevicz

Q – (comment) : We have measured 20 kV potential differences between tanker and flighter before and after refuelling.

PROTECTION/HARDENING OF AIRCRAFT ELECTRONIC SYSTEMS AGAINST THE INDIRECT EFFECTS OF LIGHTNING

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SUMMARY

This paper discusses a number of different approaches which can be applied to protect or harden aircraft and their electronic systems against the indirect effects of lightning. The basic approaches include (1) hardening the external structure to prevent or greatly reduce the penetration of rf energy into the aircraft, (2) shielding equipment and cables, (3) hardening electronic circuits, and (4) combinations of all three. The use of filters, limiters, circuit design and functional hardening are described as techniques which can be applied for protecting electronic circuits. A systems approach is recommended for achieving an optimum hardened configuration.

1. INTRODUCTION

In a previous paper presented at this Conference (Ref 1), a general approach and a modified approach for assessing the vulnerability of aircraft electronic systems to the indirect effects of lightning were outlined. In following either approach, if safety margins are found to be negative or inadequate as a result of the vulnerability assessment, then hardening measures are required. This paper identifies hardening measures or options that are available and can be implemented to achieve a hardened system - one that is adequately protected against the induced electromagnetic effects of lightning.

Four basic approaches have been suggested and applied to harden aircraft and their electronic systems against nuclear electromagnetic pulse (NEMP) effects. These approaches are generally applicable to the indirect effects of lightning. The approaches are:

(1) Harden the external structure to prevent or greatly reduce the penetration of rf energy into the aircraft.

(2) Shield equipment and cables to prevent or greatly reduce the penetration of rf energy through the shielded enclosure.

(3) Harden electronic circuits to withstand the signals that do penetrate.

(4) Combine (1), (2), and/or (3) above.

2. HARDENING THE EXTERNAL STRUCTURE

Approach (1) has been applied to a number of communication aircraft. In hardening these aircraft, it was determined that it would be most cost-effective to begin by closing the points-of-entry (POEs) where fields or currents enter from the outside. Energy which is conducted inside on cables or which propagates through the holes or apertures in the structure can result in other currents being induced on cables inside. These currents can further propagate along the cables to critical or sensitive electronic systems where permanent damage or transient upset can result. Fast digital circuits are the most susceptible to upset.

To determine POEs, different tests were run on the aircraft. The first test involved a visual inspection and an rf "sniff test" in which an rf source was placed inside the aircraft and a detector was used outside the aircraft to sense points of leakage. Additional tests involved injecting currents onto the exterior of the fuselage and measuring cable currents inside as the various POEs were closed.

Important POEs that were found included cockpit windows, cables in various locations (e.g., wings, empennage, radome, nose wheel well), various doors (e.g., engine nacelle, personnel access, avionics bay, cargo loading), power return via structure, coaxial cable shield bonding, de-icing lines, and mechanical shafts and control lines.

After important sources of rf coupling were found, various techniques were developed to close the POEs. Some of these were (Ref 2):

(1) The use of fine wire mesh in cockpit and other windows.

(2) The use of rf gaskets and special corrosion protection techniques around doors and access covers.

(3) The use of filters on cables that go outside the fuselage.

(4) The use of improved bonding on coax shields, waveguide, hydraulic lines, pneumatic lines and some mechanical shafts.

(5) The use of ferrite cores around control lines and some mechanical shafts to absorb and reflect transients.

(6) The use of a separate wire return for the neutral on power lines rather than using the structure.

(7) The use of shielding on some exposed cables in the wings, wheel wells, empennage and under the nose radome.

Using various simulated closure techniques, the average transient current on all cables inside the fuselage was reduced about 20 dB below the unhardened condition.

Approach (1) is advantageous in that it uses the existing structure to protect a large number of electronic packages that would be expensive to modify to withstand large lightning induced transients. If needed, filters, surge protection devices, coupling transformers and other hardening devices can be made smaller, lighter, and less expensive. However, closing the POEs adds some weight, particularly in the filter packages for cabling coming into the fuselage from the wings and empennage. Also, reliability and maintainability of hardening techniques have yet to be determined over the extended life of the aircraft. For example, metal bellows used for improved bonding of rudder and elevator push-pull shafts are guaranteed for 50,000 full-stroke cycles but should last much longer for smaller stroke (normal) operation. The only limited life items are the door seals which will have a wearout problem, but which can be easily replaced.

Recently, studies of the degradation of some of these hardening techniques (i.e., filtering, cable shields and conductive gaskets) were reported (Ref 3). The shielding effectiveness of double shielded cables was measured both before and after the cables were subjected to a series of bending, twisting, and stretching tests (500 bends, 500 rotations). The shielding effectiveness was found to degrade from an expected 60 dB to about 30dB in some cases. Measurements on filter packs in systems subject to field environments showed that filters which had been expected to provide 50 dB of attenuation provided as little as 22 dB due principally to improper grounding or installation. Conductive gaskets were measured under varying pressures to predict EM degradation with age since they are known to take a set when compressed for a long period of time.

3. SHIELDING EQUIPMENT AND CABLES.

Approach (2) was employed in shielding electronic system elements, known as Line Replaceable Units (LRU's), in the B-1 aircraft against the nuclear electromagnetic pulse (Ref 4). As far as practicable, LRU's were grouped in a relatively few avionics bays which were designed and constructed to give 70 dB attenuation against the external NEMP environment over a specified frequency range. Within the bays, further shielding was not necessary. In the design and construction of the bays, special attention was given to joint fabrication, door seals, and to intrusion control. Electrical interconnections between bays were made principally by high-permeability conduits. In high vibration areas, and for ease of manufacture, braided-shield cables were used extensively. Special care was exercised in shielding and terminating the cables. The remainder of the aircraft was protected by the basic structure of the aircraft which was conservatively considered to be 20 dB.

The following guidelines have been recommended when shielding all the cables and electronic packages within the aircraft (Ref 5):

(1) Use a one-point circuitry ground.

(2) Separate all power leads.

(3) Balance power leads on all cable runs.

(4) Use the shielded twisted pair wiring concept on all lines except power lines.

(5) Build all power leads in separate bulk shielded cables.

(6) Build all other leads in bulk shielded cables.

(7) Use metallic enclosures for all electrical and electronic packages.

(8) Filter all leads entering or leaving high noise level packages. This includes motors, choppers, inverters, power relays, high power transitor switching circuits, etc.

(9) Filter all leads entering or leaving a transmitter package except the transmitter output coax.

(10) Use a filter/arc suppressor on all deliberate receiving antennas and on low power transmitters using semiconductor output devices. Filtering will be limited by pass band requirements, but the combined filter/arc suppression will, in general, handle the problem.

4. HARDENING ELECTRONIC CIRCUITS

Techniques which can be employed in Approach (3) to harden electronic circuits to withstand transients that penetrate the external structure and shielded equipment bays and cable runs include filtering, limiting, circuit design, and functional hardening. Each of these techniques are discussed below.

4.1 Filtering

A filter provides a reduction in power to a load of undesired signals while permitting desired signals to pass with little or no attenuation. The filter accomplishes this by discriminating against the frequency content of the undesired signals. However, if the interfering signal has high content in the desired (pass) band, the filter effectiveness is reduced.

A filter must meet some or all of the following requirements to be effective:

(1) Have extremely high or low input impedance relative to the source in the frequency rejection (stop) band.

(2) Have high loss to frequencies in the stop band.

(3) Have high dc current carrying capability without changing stop band characteristics.

(4) Prevent cross coupling.

(5) Prevent arcing across its elements.

(6) Have low dc resistance (primarily power line consideration).

(7) Have high reliability.

A wide range of filter types are available for lightning electromagnetic pulse (LEMP) hardening. These include:

(1) Discrete R, L, C filters

(2) Ferrite filters

(3) Filter pin connectors

(4) Coaxial filters

(5) Electromechnical filters

(6) Active filters

Passive discrete filters use combinations of discrete R, L, and C components. They permit implementation of any of the functional filter characteristics (i.e., high pass, low pass, bandpass, and band reject). With careful attention to component selection, they can be designed for use at frequencies up to 100 MHz.

The ferrite filter is a special category of discrete passive filters. This filter achieves a frequency dependent resistive insertion loss by using a dissipative filter at the high frequencies in the stop band while avoiding the low frequency (pass band) dissipation associated with the other discrete passive dissipative filter, the RC filter.

Filter pin connectors are passive and use the coaxial shell of cable connectors as a capacitor plate along with added series inductance to obtain a network filter. They are low pass filters and provide cut off frequencies above 10 kHz.

Coaxial filters are distributive passive filters that utilize short sections of high impedance or low impedance transmission line to simulate series inductance and shunt capacitance, respectively.

Monolithic crystal and ceramic filters and mechanical filters are categorized as electromechanical filters. Crystal and ceramic filters use the piezoelectric effect to transform electrical energy to mechanical energy and then use the mechanical resonance properties of the crystal or ceramic to achieve frequency selectivity. Mechanical filters use a mechanical resonator with electromechanical transducers at the input and output to achieve electrical filter properties. These filters are basically bandpass circuits, but low and high pass functions can be achieved by adding passive elements.

Active filters, while not generally applicable to LEMP protection, may be useful in some specific applications. Active filters are hybrid devices that use one of four basically different active circuits or devices in conjunction with passive frequency discrimination elements. The four circuits include the negative impedance converter, the operational amplifier, the gyrator, and the phase-locked loop.

Filter types and their characteristics are summarized in Table 1 (Ref 6).

4.2 Limiting

Limiting is an effective method to reduce coupled energy from getting into sensitive electronic circuits. Limiting (or clamping) is achieved by using special suppression devices which are of two basic types: "crowbar" and "constant voltage." "Crowbar" devices are those which, on sensing an overvoltage, switch to a low impedance state and thus cause the impressed voltage across them to collapse to a low value. "Constant voltage" devices are those which, on sensing an overvoltage, tend by virtue of their nonlinear current-voltage relation to maintain their voltage at a predetermined level and do not collapse the voltage. Examples of the first type are spark gaps and gas filled surge arrestors.

Examples of the second type are Zeners and silicon avalanche diodes and varistors.

"Crowbar" or switching devices inherently offer greater surge power handling capability than do the Zener or varistor type of devices. The instantaneous power dissipated in a suppressor device is a product of the surge current through the device and the voltage across the device. For a constant surge current, a switching device like a spark gap which has low voltage across it when conducting will dissipate less power than a device such as a Zener diode which retains a high voltage across it. Thus, a spark gap will be physically smaller than a Zener diode or varistor device for a given surge power handling capability.

Major disadvantages of switching devices are their inability to clamp surge voltages to a low level and their tendency to continue conduction once started on dc lines of significant voltage.

The metal oxide varistor (MOV) is a "soft" limiter whose shunt resistance is inversely proportional to the applied voltage. No critical voltage level of surge is required to activate this protection device as it is always in the circuit as a path to ground. As the surge voltage rises, the varistor lowers its resistance thus providing a low shunt impedance in parallel with the device being protected. A major disadvantage of this type of device on some circuits is its loading effect on the circuit during periods of steady state operation.

Semiconductor devices such as the Zener diode and silicon junction suppressor have the ability to provide protection on voltage limiting down to low voltage levels necessary to protect integrated circuits and other semiconductor devices. Their major drawback is their limited capability to handle significant surge energy.

Table 2 lists several of the basic surge protection devices and their advantages and limitations (Ref 7).

Hybrid protection using both spark gaps and silicon avalanche suppressors have been developed for optimizing the capabilities of both of the device types. The spark gap contributes the high current surge capability while the silicon component responds to the fast rise time wave front components.

4.3 Circuit Design

An important means of controlling lightning-related interference is through proper circuit design. Basic considerations about circuit design and signal transmission are shown in Figure 1 (Ref 8).

As shown in Figure 1(a), signal circuits should avoid the use of the aircraft structure as a return path. If the structure is used as a return path, resistively and inductively generated voltage drops will be included in the path between transmitting and receiving devices.

Signal transmission over a twisted pair circuit as shown in Figure 1(b) with signal grounds isolated from the aircraft structure tends to couple lower voltages in the signal path. However, the use of twisted pair transmission lines does not eliminate the common mode voltage to which electronic systems may be subjected. Common mode voltages applied to the unbalanced transmission path can lead to line-to-line voltages comparable to the common mode voltage.

For differential transmission and reception devices shown in Figure 1(c), a many-fold improvement can be achieved in rejecting common mode voltages produced by lightning.

In general, it is good practice not to interconnect two different pieces of electronic equipment at semiconductor junctions as shown in Figure 1(a) and 1(c). Resistance inserted between the junction and the interconnecting wires as shown in Figure 1(d) can dissipate transient energy and effectively protect semiconductors from lightning induced voltages and currents.

Finally, interconnection through balanced transmission lines and transformers as shown in Figure 1(e) in conjunction with input protection for semiconductors probably provides maximum protection against unwanted transients induced on signal wiring.

4.4 Functional Hardening

Functional hardening is particularly important in protecting digital circuits which have broad bandpass characteristics and which are most susceptible to system upset. From a system standpoint, functional hardening can be pursued using several different approaches (Ref 4).

Use of a higher operating voltage level for digital circuits probably obtains more protection than is possible by any other means. If digital data are transmitted over a cabling system at 10 volts, for example, a 60 dB advantage results from a protection standpoint than if a 10 millivolt drive level were chosen.

Coding and signal processing techniques can be employed to minimize the possibility of operational upset. Some of the techniques include error detection and correction codes, repeated data, parity checks~~, closed loop informa~~tion transfer, and detection reject circuits.

Hard memories can be used as temporary storage in critical applications where long computational processes are required to generate needed data. This technique can obviate complete recycling of computer programs should some form of operational upset occur.

Circumvention techniques can be used when hardening requirements are physically difficult to apply. For a specific threat, circumvention involves detecting the event and using this information to blank or reset a computational process. For non-specific threats, a duty cycle technique is used whereby information is accepted only during small increments of time so that the probability of a lightning event, for example, happening during this time period is negligible or extremely small. Event sensors can also be used to reset systems or initiate master resets.

The use of fiber optics in signal circuits offers the possibility of practically eliminating electromagnetic interference and transient pulse effects. However, fiber optics is presently limited to signal circuits and will not affect the problem of transients induced via power circuits.

5. CONCLUSIONS AND RECOMMENDATIONS

This paper has touched briefly upon a number of approaches and techniques that have proved effective in protecting aircraft electronic circuits from the induced electromagnetic effects of both NEMP and lightning. To be most effective, a systems approach is required to insure that no one system is either overly hardened or underprotected. With proper analysis and testing, a balance of hardening options can be chosen and applied to achieve an optimum hardened configuration.

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TABLE 1. FILTER COMPARISON MATRIX

FILTER CLASS	FILTER TYPE *	USEFUL FREQUENCY RANGE (Hz)	SIGNIFICANT ADVANTAGES	SIGNIFICANT DISADVANTAGES
Discrete R, L, C	1, 2, 3, 4	to 10^8	Versatile Low Cost	Large for Low Frequency Low Q
Ferrite Beads	1	$10^6 - 10^8$	Versatile Dissipative With Low Pass Band Loss	Spurious Resonances Saturation
Filter Connector	1	$10^4 - 10^9$	Design Integration Simplicity Dissipative	Spurious Resonances Saturation
Coaxial	1, 2, 3, 4	$10^7 - 10^9$	High Frequency Use Low Parasitics	Large Size
Crystal	3, 4,	$5x10^6 - 1.5x10^8$	High Q Small Size	Spurious Resonances High Cost
Ceramic	3	$10^5 - 10^7$	High Q Small Size	Spurious Resonances Not IC Compatible
Mechanical	3, 4	$.1 - 2x10^4$	High Q	Limited Range Not IC Compatible High Insertion Loss
Active	1, 2, 3, 4	to 10^5	Small Size Gain Provision	Power Requirement Limited Range Damage Susceptibility

* 1 - Low pass; 2 - High Pass; 3 - Bandpass; 4 - Band Reject

TABLE 2

COMPARATIVE VALUES OF SEVERAL SURGE PROTECTION DEVICES

PARAMETER	SPARK GAPS	VARISTOR DEVICES	ZENER DIODES
Typical surge current capability (Amps)	10,000	1,000	500
Response Time (Sec)	10^{-8}	10^{-8}	10^{-9}
Capacitance (fd)	10^{-12}	10^{-10}	10^{-10}
Voltage Range (Volts)	90 and higher	40-700	2-300
Insulation Resistance	High (10^9 ohm)	Medium	Medium
Bi-Polar Operation	Yes	Yes	No
Failure Mode	Short	Short	Short
Activated State	Short Circuit	Clamped	Clamped

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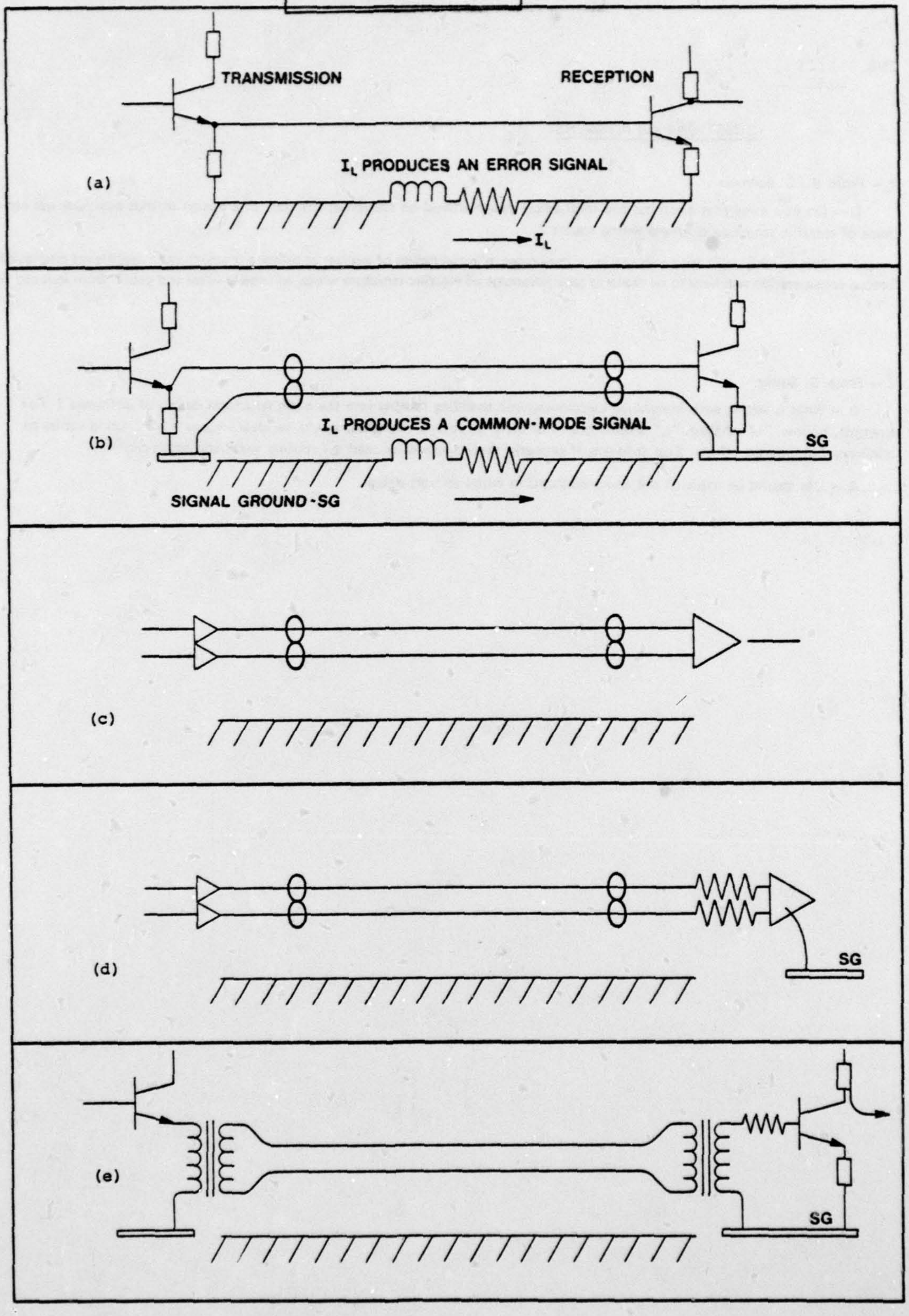

FIGURE 1 - CONSIDERATIONS REGARDING CIRCUIT DESIGN

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QUESTIONS and ANSWERS

1 – From B.J.C. Burrows

Q – Do you agree that electrical and mechanical design should be treated at the start of a design so that optimum use can be made of metallic structure to shield wiring routes ?

A – Wire routing will become increasingly important in initial design of aircraft as composite structural materials are employed. Special consideration will have to be made to take advantage of metallic structure which will shield wires and cables from induced fields.

2 – From D. Suiter

Q – What is wrong with integrating electromagnetic shielding designs into the basic structural design of airframes ? For example, hollow, "U" shaped, "L" shaped beams or other structural members, would be ideal locations for routing cables to minimize EM induced effects. Skin stringers, if properly shaped, could be used for routing wires and small cables.

A – Use should be made of the above concepts in initial aircraft design.

STATIC ELECTRICITY PHENOMENA: THEORY AND PROBLEMS

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Static electrification of aircraft and insulating surfaces on aircraft causes a variety of problems ranging from radio noise generation to personnel shock and explosive initiation. The charging mechanisms responsible for aircraft electrification are discussed together with the problems they cause.

I INTRODUCTION

A. General

Static electrification occurs whenever two materials are brought into contact and separated.[1] Electric charges remain on the surface after the separation of solid-to-solid contacts. The two materials involved may be a combination of conductors, semiconductors, and insulators. In the applications of interest here, there is a flow of material involved--ice crystals striking an aircraft, or fuel flowing in a pipe--so that, in time, substantial quantities of charge can be separated, and high voltages developed.

Frictional electrification involves two equally important physical phenomena: (1) electrification of the interface, and (2) charge backflow. The first phenomenon may be studied in light of solid state physics. Only recently have efforts along these lines been successful and in these, the experimenters worked with freshly-cleaned surfaces in hard vacuum systems.[2] In practice, it is necessary to deal with complex materials and surfaces having substantial contamination. Humidity, temperature, dust, gaseous pollutants, external fields, etc. can have a considerable influence on both charge polarity and magnitude. For example, as little as mono-molecular layer of adsorbed gas may sufficiently change the work function of a solid material to reverse the polarity of the resulting electrification.

The second phenomenon, charge backflow, is highly important in determining the net charge separation since the amount of charge that flows back during separation may be an order of magnitude, or more, greater than the charge which remains after contact and separation are completed. It is believed that the charge backflow depends on several parameters, such as the types of materials in contact, the surface preparation, the geometry of the contact, the speed of separation, the temperature, the ambient gas, etc.

In view of the critical dependence of static electrification on poorly-understood processes and uncontrolled parameters, it follows that specific engineering applications involving static electrification must be based on experimental results of a sufficiently large number of tests, as well as on the proven ability to maintain or reproduce the surface and ambient conditions under which the original tests were carried out. Accordingly, static electrification processes will be considered here primarily in terms of the charging magnitudes involved and the effects they produce on aircraft systems.

B. Motivation for Study of Static Electricity

Static electricity causes a variety of problems associated with aircraft operation. One of the first to be recognized was "precipitation static"--the noise observed on radio systems whenever an aircraft became electrically charged upon flying into precipitation containing ice crystals.[3-5] With the advent of helicopters, it was also found that, as a result of static charge on the helicopter, ground cargo handlers may receive severe shocks upon touching lowered cargo hooks.[6,7] In addition, ground crews are often shocked if they touch charged windshields shortly after an aircraft lands.

Fuel being pumped either on the ground or in flight becomes charged, and can cause explosion hazards unless proper precautions are taken.[8-10] It has also been found that static charges accumulated on dielectric surfaces can fire electroexplosive devices on military aircraft.[11,12]

II STATIC CHARGING IN FLIGHT

A. General

The various ways in which static electrification of a conventional aircraft can occur are illustrated in Figure 1. Figure 1(a) illustrates frictional electrification; as uncharged precipitation particles strike the aircraft, they acquire a positive charge, leaving an equal and opposite negative charge on the aircraft and raising its potential to tens or hundred of thousands of volts.[13,16] Charging occurs both on the metal structure of the aircraft and on dielectric surfaces such as the windshield. Dielectric surfaces can thus become charged with respect to the airframe.[17,18] Engine charging, illustrated in Figure 1(b), occurs when flight vehicles are operated at low altitudes.[13,14] Processes as yet incompletely understood occur within the engine combustion chamber and cause a

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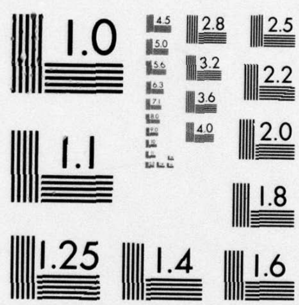

MICROCOPY RESOLUTION TEST CHART
NATIONAL BUREAU OF STANDARDS-1963-A

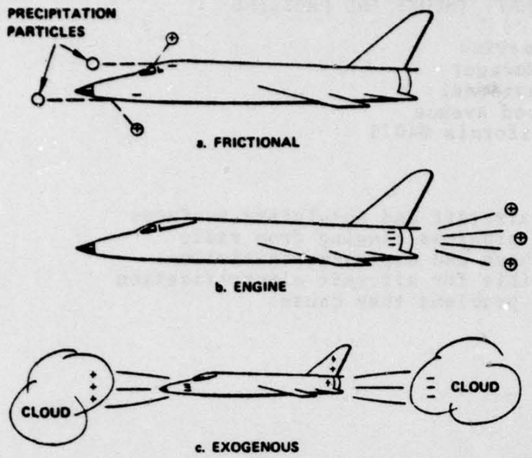

FIGURE 1 CHARGING PROCESSES

predominantly positive charge to be expelled with the engine exhaust. This causes an equal and opposite (negative) charge to be imparted to the aircraft charging it to potentials of tens or hundreds of thousands of volts. Exogenous charging, illustrated in Figure 1(c), occurs when the vehicle flies in a region of electric field, such as that generated between oppositely charged regions of clouds; this field can cause discharges to occur from the extremeties of the vehicle.

The operational conditions under which static electrification can occur depend somewhat on the class of vehicle. Since airplanes encounter severe charging during operation in clouds in horizontal flight, electrification can continue for considerable periods of time on all-weather missions. On jet aircraft operating at low altitude, engine charging can be an additional source of long-term electrification. Helicopters become charged while flying through naturally occurring clouds. In addition, a hovering helicopter can stir up snow or dust thereby generating its own cloud of particles to produce frictional electrification. Thus, helicopters encounter static problems in regions where conventional aircraft do not.

The charging process itself produces virtually no difficulty, but vehicle voltage and electric fields can become so high after a period of time that electrical discharges occur. It is the discharge of the accumulated static electricity that generally produces the most harmful effects.

An important consequence of static electrification is electrical noise. The various noise mechanisms that have been identified are shown in Figure 2. As the airplane becomes charged, the electric fields at the extremeties of the vehicle become sufficiently high to cause corona breakdown of the air.[13,15] At the operating altitude of airplanes, this breakdown occurs as a series of very short pulses containing energy in the radio frequency spectrum. These noise pulses can couple into communication, navigation, or digital circuitry to produce interference.

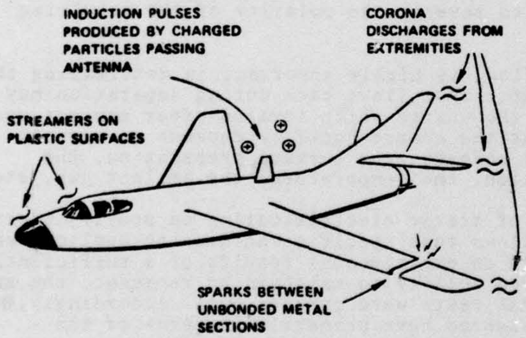

FIGURE 2 NOISE SOURCES

Another source of noise occurs when dielectric surfaces on the front of the airplane, such as the windshield and radome, are exposed to frictional charging, as illustrated in Figure 2. These surfaces can be charged by impinging particles. Since these materials are insulators, the charge is bound at the place where it was deposited and cannot be discharged until sufficient electric charge has accumulated to produce a streamer (a spark-like discharge) across the dielectric surface to the metal airframe.[17,18] Streamer discharges are short in duration, and involve the transport of charge over a large distance. They therefore produce radio frequency interference which can couple into susceptible systems on the aircraft. In some cases, the streamering on a square inch of surface in a critical location is sufficient to disable systems.

A third source of interference that often occurs inadvertently on airplanes is associated with sparking between unbonded adjacent metal sections of the aircraft.[14] For example, consider Figure 2, which shows a break in the wing; charging processes on the airframe will raise the potential of the inboard section with respect to the outboard section until a spark occurs in the gap. This spark produces a short current pulse, which is also a source of noise. In flight, the current required for corona discharge from the isolated wing tip is supplied from the remainder of the airplane.

Finally, slowly varying induction pulses can be produced in antennas by the passage of charged particles.[17] This noise is of importance only at VLF or ELF and does not pose much of a problem to conventional communication and navigation equipment. With the advent of systems operating at frequencies of the order of 10 kHz, however, induction noise should be considered.

B. Airplane Charging

Although the three mechanisms shown in Figure 1 can produce aircraft charging, the frictional charging process is of most concern for the following reasons. First, flight test measurements indicate that the charging currents produced by frictional charging during flight through ice crystal clouds are substantially higher than the currents

resulting from engine charging. Thus, aside from the fact that charging occurs during flight through clear air, no new problems are caused by engine charging. Second, the amount of time an aircraft spends in the vicinity of electrically charged clouds as shown in Figure 1(c) is much less than the time it spends flying through uncharged precipitation.

The precipitation charging current to a vehicle is given by

$$i = q_p \, c \, v \, A_{eff}$$

where

q_p = Charge per particle
c = Particle concentration
v = Aircraft velocity
A_{eff} = Effective intercepting area of aircraft.

The various parameters in the equation and their interdependencies have been studied analytically, in the laboratory and in flight, and are generally understood for the operating regimes of current aircraft.[13,15] Typical values of particle parameters for an aircraft operating in the subsonic flight regime are given in Table I for two cloud types.

Table I Precipitation Particle Parameters

Cloud Type	q_p pico Coulomb	c m^{-3}
Cirrus	1 - 10	2×10^4
Thunderstorm Anvil	1 - 35	5×10^4

Laboratory experiments involving the charging of projectiles fired through ice crystal clouds were conducted to determine the relationship between the charge acquired and the impact velocity.[19] The results of these experiments indicate that the projectile charge decreases with increasing velocity as shown in Figure 3. These results were further verified by flight tests.[20] It was noted that the observed effect might be caused by the melting of the ice crystals by the energy of the impact, since flight-test experience indicates that clouds composed of water droplets tend to charge an aircraft at a much lower rate than do clouds containing ice crystals. Thus, if an ice crystal is completely melted upon impact, greatly reduced charging would result.

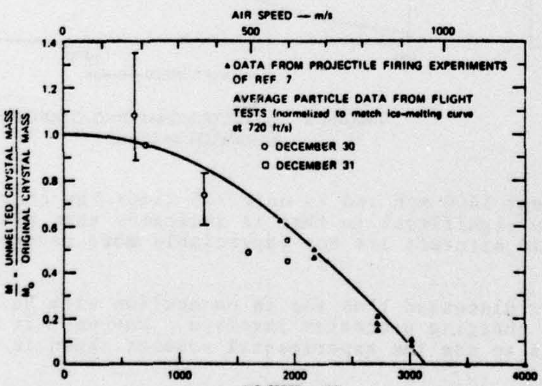

FIGURE 3 VARIATION OF PARTICLE CHARGE WITH SPEED

The maximum possible heat input to an ice crystal at the time of impact is given by the kinetic energy of motion, $KE = 1/2 \, Mv^2$, where M is the ice crystal mass and v is the velocity of the crystal with respect to the projectile. Assuming an initial temperature of $-55^{o}C$, an ice crystal can be completely melted by the kinetic energy of an impact at velocities in excess of 3150 ft/s. At reduced speeds, only part of the ice crystal would be melted. If the charge acquired by an ice crystal is proportional to the mass of the ice it contains, one can argue that the particle charge will be proportional to the unmelted ice mass remaining after an impact.

A curve of unmelted ice-crystal mass as a function of velocity (assuming that all kinetic energy is transformed to thermal energy) is plotted in Figure 3. Average values of charging measured during the projectile firing experiments (normalized to fit the ice-melting curve at 1870 ft/s) are also plotted as triangles in the figure. The fact that the charging decreases with increasing speed in the same general manner as the unmelted ice mass, and the fact that charging vanishes at roughly the speed required to permit the crystal to melt, indicate that melting is most likely responsible for the reduced charging resulting from ice-crystal impacts at supersonic speeds. The flight test data normalized to fall on the ice-melting curve at a speed of 718 ft/s are also plotted on Figure 3. It is evident that the individual particle charge measured in flight does tend to decrease with increasing speed in the same general way as did the laboratory data. There is considerable scatter in the individual data points--somewhat more than was observed in the laboratory--but the flight-test data follow the trend of the ice-melting curve.

Aircraft icing studies indicate that the effective intercepting area A_{eff} in Eq. (1) is affected by aircraft speed as illustrated in Figure 4.[21-25] The particles tend to be swept around the aircraft by the windstream so that only the particles in the shaded region actually strike the aircraft. As speed is increased, particle inertia causes them to deviate more from the air streamlines and the effective intercepting area more nearly approaches the projected frontal area.

For a given speed and body shape, the fraction of the projected frontal area effective in intercepting particles depends on body size. A small aircraft is more efficient in intercepting particles than is a large aircraft. Also, the airfoils are more efficient interceptors than the fuselage.

The results of studies of water droplet impingement on airfoils indicate that the effective intercepting area of a typical aircraft would vary with speed as shown in Figure 5.[19] Combining the results of Figures 3 and 5 yields the curve of Figure 6 which indicates the predicted charging current behavior as a function of speed.[19] It is noted that because of ice-crystal melting, the charging rate decreases rapidly at speeds above 1500 mph.

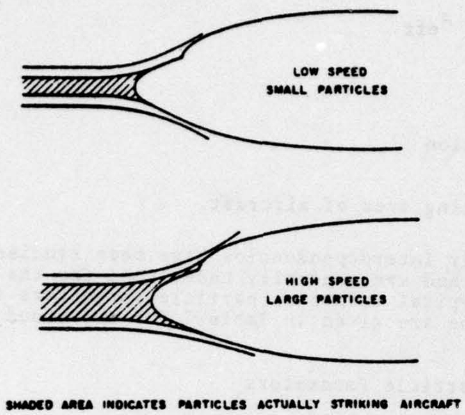

FIGURE 4 EFFECT OF AIRPLANE SPEED ON PARTICLE INTERCEPTING AREA

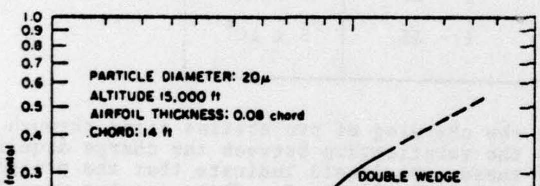

FIGURE 5 ADVANCED AIRCRAFT EFFECT INTERCEPTING AREA

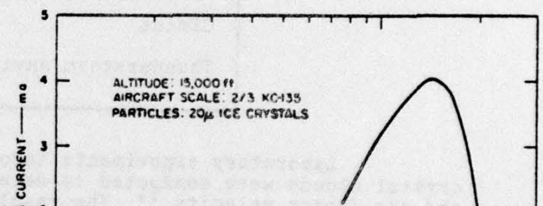

FIGURE 6 PREDICTED CHARGING CURRENT FOR ADVANCED AIRCRAFT

The maximum charging current occurs at about 1400 mph and is only 2.6 times the charging current at 600 mph. This result is highly significant in that it indicates that precipitation static problems on highly supersonic aircraft are not appreciably more severe than they are on subsonic aircraft.

The studies of aircraft charging discussed thus far in connection with Eq. (1) provide important insights into the basic charging processes involved. However, it is simpler in practical charging calculations to use the experimental results shown in Table II directly.[13,15]

Table II

Peak Charging Rates Encountered With KC-135 Prototype

Cloud Type	Peak Charging Rate ρ (μamp/sq ft)
Cirrus	5 to 10
Strato Cumulus	10 to 20
Frontal Snow	30

These are the peak charging rates encountered during instrumented flight tests conducted using the KC-135 prototype aircraft. The data should be typical for subsonic jet aircraft operating at normal cruising speeds.

C. Helicopter Charging and Shock Problems

Figure 7 shows a record of charging current and helicopter potential measured on a Chinook helicopter hovering in a dusty environment. T = 0 denotes roughly where

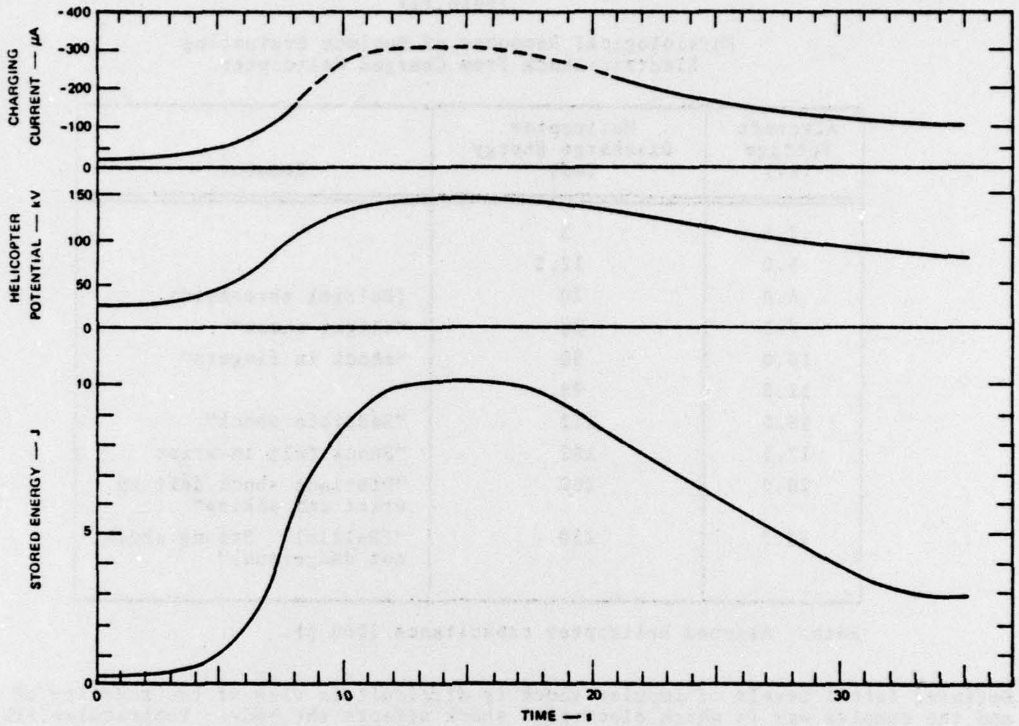

FIGURE 7 HELICOPTER POTENTIAL AND CHARGING CURRENT AS A FUNCTION OF TIME AFTER LIFTOFF IN A DUSTY ENVIRONMENT

take-off occurred and the hover altitude of 50 ft over the ground array was established at approximately 15 seconds.[26] In this figure the charging current rapidly increases to about 300 μA and then at about t = 17 seconds slowly begins to decay. This decay is due to the rotor wash blowing the dust away from the measurement site. It should be noted that the helicopter potential reached a maximum value of 140 kV and then gradually decayed to roughly 75 kV. Since the self capacitance of a hovering Chinook helicopter is roughly 1000 pF, these potentials correspond to stored energies of 10 Joules and 3 Joules respectively.

Discussing helicopter potentials and stored energy is not meaningful unless it can be related to a person's response to these quantities. To address this question in a simulated environment, the test helicopter was configured as shown in Figure 8 and hovered

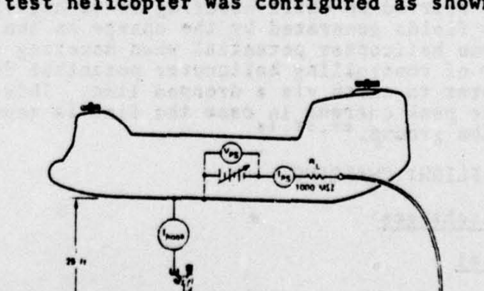

FIGURE 8 AIRCRAFT CONFIGURATION USED TO MEASURE PHYSIOLOGICAL RESPONSE TO ELECTRIC SHOCK

25 ft over a concrete pad.[27] This test was conducted by raising the helicopter up to some potential, V, using the internal power supply. The (barefooted) person beneath the helicopter then touched the simulated cargo hook and noted his response. Helicopter voltages, discharge energies, and the subject's observations about the shock levels experienced are shown in Table III. It can be seen from this table that the subject's comments agree well with data published by Dalziel.[28,29] The data in Table III also demonstrate that the tolerable helicopter voltage is substantially below the helicopter potentials measured in Figure 7.

Table III

Physiological Response of Subject Evaluating Electric-Shock From Charged Helicopter

Aircraft Voltage (kV)	Helicopter Discharge Energy (mJ)	Comments
2.5	3	
5.0	12.5	
6.0	20	(Dalziel threshold)
7.5	28	"Slight shock"
10.0	50	"shock in fingers"
12.5	78	
15.0	112	"Sensible shock"
17.5	152	"Shock felt in wrist"
20.0	200	"Distinct shock felt in wrist and ankles"
23.7	250	"(Dalziel: Strong shock, not dangerous)"

Note: Assumed helicopter capacitance 1000 pF.

Defining lethal levels of impulse shock is difficult in view of the scarcity of data, and the complex way in which electrical shock affects the body. Ventricular fibrillation, the most common cause of death in electric shock cases, occurs at intermediate levels of current. Below this level, only unpleasant sensations are felt, while very strong short shocks will stop a fibrillating heart. Dalziel describes a case in Sweden in which a 22 year old man was killed as the result of receiving the residual charge from the capacitors of a high voltage filter supplying a 150 kW transmitter. The energy received by the victim was estimated to be 24 Joules. The same paper discusses other cases in which people received comparable or higher shocks and survived with burns, unconsciousness, or severe headaches. The conservative interpretation of Dalziel's data is that the peak potentials and stored energies in Figure 7 are not acceptable for cargo handling operations, and that means for controlling the helicopter potential must be provided.

Various efforts have been made to devise active discharge systems to sense the helicopter-to-ground potential and then to discharge current of the appropriate polarity to maintain the helicopter potential within acceptable limits.[26,31-34] The proposed systems have all relied upon the use of one or more static electric field meters mounted on the helicopter to determine its potential. Unfortunately, recent flight tests indicate that the dust cloud stirred up while hovering is so highly charged that its fields completely mask the fields generated by the charge on the helicopter making it impossible to determine the true helicopter potential when hovering in dust.[26] At present, it appears that the best means of controlling helicopter potential during cargo handling is to connect the helicopter to earth via a dropped line. This line must include a resistive element to limit the peak current in case the line is touched by the cargo handlers before it reaches the ground.[27,35,36]

III EFFECTS OF IN-FLIGHT CHARGING

A. Corona Discharges

1. General

As was indicated earlier, when the overall aircraft potential is increased a threshold will be reached above which corona discharge breakdown occurs in regions of high electric field at the aircraft extremeties. If the potential is further raised, the discharge current will increase as the field over more and more of the extremety exceeds the threshold value. The locations of the corona discharges are determined by the gross geometry of the aircraft, which determines the gross field structure and by the presence of sharp points which produce localized field concentrations.

2. Corona Discharge Characteristics

Individual pulses associated with a corona discharge have the general form indicated in Figure 9.[37] The precise amplitude and time structure are a function of aircraft altitude and discharge point radius. Laboratory measurements on a corona discharge from a sheared metal edge (simulating the trailing edge of an airfoil) indicate that, at atmospheric pressure, the pulse repetition frequency (prf) is on the order of 10^5 pulses per second when the discharge current is 100 μA.[13] Thus each pulse carries away 10^{-9} Coul. of charge. Since a precipitation particle deposits roughly 10^{-11} Coul. of charge, each corona pulse removes the charge deposited by 100 precipitation particles. Since the capacitance of a large aircraft such as the 707 is 100 pF, the charge carried away by a single corona pulse changes the aircraft potential by 1 volt.

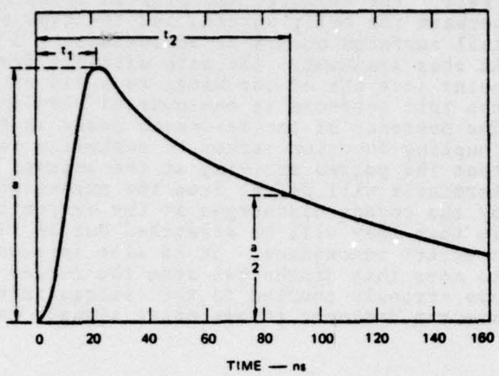

FIGURE 9 TIME STRUCTURE OF NEGATIVE POINT CORONA PULSE, PRESSURE = 200 mm/Hg

Although data regarding corona pulse form and other characteristics provide insight into the corona discharge process, corona noise calculations are best carried out using noise spectral data of the sort shown in Figure 10.[13,15] These data present measured spectral characteristics for the entire aircraft operating altitude range. It should be noted that the corona source is most energetic at low frequencies, but that it contains appreciable energy well into the HF band.

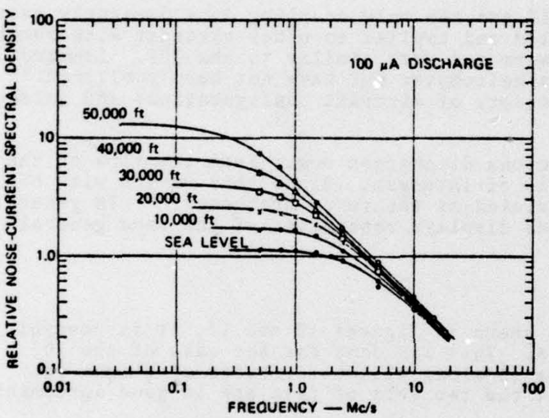

FIGURE 10 NORMALIZED NOISE SPECTRUM GENERATED BY CORONA DISCHARGE FROM TRAILING EDGE

When the source characterization work leading to Figure 10 was under way, aircraft used the VHF and UHF bands strictly for line-of-sight communication and navigational aids operating at received signal levels sufficiently high that corona noise at these frequencies was not of concern. Accordingly no effort was made to extend the corona spectrum studies above 14 MHz. This is unfortunate since it is now planned that satellite communication links operating at low received signal levels will be employed on aircraft. To accurately assess their vulnerability, it would be useful to have corona spectral data extending to VHF and UHF.

3. Coupling of Corona Noise

In general, the corona noise source on an aircraft is located in one place, and the affected antenna or system is located in another. It is therefore necessary to define the coupling between the noise source and the victim system. This was done for two antenna locations on the 707 aircraft as illustrated in Figure 11.[13,15] The antennas used in making the measurements were a small tail-cap and a flush belly antenna located in a fairing at the root of the wing. Coupling was measured between each of these antennas and the noise source regions at each of the airfoil extremeties. The results of the measurements are shown in Figure 11.

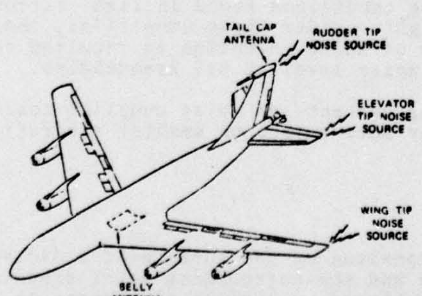

FIGURE 11 ANTENNA AND CORONA-NOISE-SOURCE LOCATIONS FOR 707 COUPLING MEASUREMENTS

Although the coupling-measurement technique used inevitably includes the test-antenna characteristics in the measured results, the form of the coupling as a function of frequency is not affected by the details of the test antenna, provided the antenna dimensions are small compared to a wavelength at the frequencies of interest. In this case, only the magnitude of the coupling function will be affected by a change in the antenna. The data shown in Figure 12 have therefore been adjusted to represent the coupling to an antenna having unity induction area, a, in response to a low-frequency, vertically polarized signal.

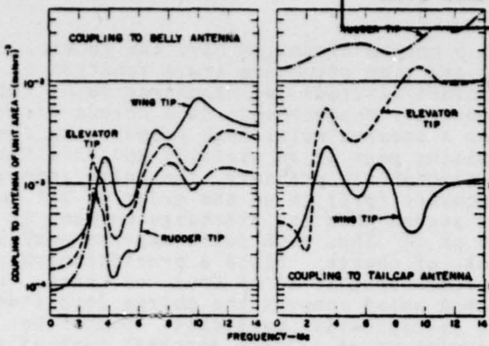

FIGURE 12 MEASURED COUPLING TO ANTENNAS ON BOEING 707 PROTOTYPE

The variation with frequency of the coupling to the various points is of considerable interest, since it shows the effect of the various electromagnetic resonances of the aircraft. For example, the peak in coupling between the belly antenna and the tips of the tail surfaces occurs at approximately 3 MHz. At this frequency, the path distance from a point just aft of the wings to a tip of one of the tail surfaces is one-quarter wavelength. The presence of the resonance peaks in the coupling function serves to emphasize the fact that the pulses arriving at the antenna terminals will differ from the pulses generated by the corona discharges at the extremities in that they will be stretched out in time by aircraft resonances. It is also interesting to note that discharges from the rudder tip are strongly coupled to the tailcap antenna. Discharges from the fin will therefore generate the dominant corona noise signals in the tailcap.

The data of Figure 12 can be applied without restriction to aircraft of similar shape but different size. In general, the same resonances will occur on the scaled aircraft, but they will occur at the scaled frequency. The magnitude of the coupling must also be scaled with aircraft size.[13,15]

Actually, the data of Figure 12 are the only coupling data presently available for conventional aircraft and have been scaled and applied to other aircraft with remarkably good results even when the scaled aircraft were not very similar to the 707. Limited coupling studies have been carried out for a helicopter but have not been published.[38] Additional coupling data covering a wider variety of aircraft configurations and antenna locations are badly needed.

In the case of helicopters, corona discharges occur from the tips of the blades, so that coupling to the blade tips is of interest. Since this varies with blade position, the corona noise intensity is modulated at the rotor frequency.[38] In general, the coupling varies with antenna location and displays resonances of the same general sort observed in Figure 12.

4. Corona Noise Levels

By combining data of the sort shown in Figures 10 and 12, it is possible to predict the corona discharge noise levels. This was done for the case of the 707 aircraft and the results are shown in Figure 13 along with the results of flight test noise measurements.[13,15] It is evident that the two sets of data are in good agreement for both antennas.

The corona noise level in the tailcap antenna is substantially above the accepted value of nighttime atmospheric noise.[39] Since the system designer can achieve improved performance until he reduces his system input-noise figure to the atmospheric noise level, it must be assumed that optimized systems are now or will be operating at the atmospheric noise level. Thus the tailcap corona noise severely degrades the performance of such a system. It should also be observed that 40 dB of noise reduction is required to reduce corona noise to the published nighttime atmospheric level. Even greater noise reduction is required to approach daytime atmospheric levels.

Although the belly antenna noise levels shown in Figure 13 are comparable to the nighttime atmospheric noise at certain frequencies, it is important to note that a 350 μA charging current is typical of the charging conditions found in light cirrus, and that currents up to 3 MA were measured in flight. Under these conditions, the noise levels will be 10 dB higher. Thus at least 30 dB of noise reduction is required to reduce corona noise to the nighttime atmospheric noise level at all frequencies.

Because of the way in which charging current and noise coupling scale with aircraft size, corona noise problems are generally more severe on smaller aircraft.

B. Dielectric Surface Discharges

1. Streamer Discharge Characteristics

As was indicated earlier, charge deposited on the surface of a dielectric causes a potential difference between the surface and the surrounding metal structures. This potential difference is usually eventually relieved by surface discharges or, streamers, originating on the rim and extending out on the dielectric surface. Typical current pulses induced on a wire beneath the dielectric on which the discharges occur are shown in Figure 14.[17,18] It is evident from Figure 14 that the time waveform of the streamer pulse depends on the length of the streamer. The amplitude of the induced pulse, of course, depends on the coupling between the streamer and the receiving "antenna."

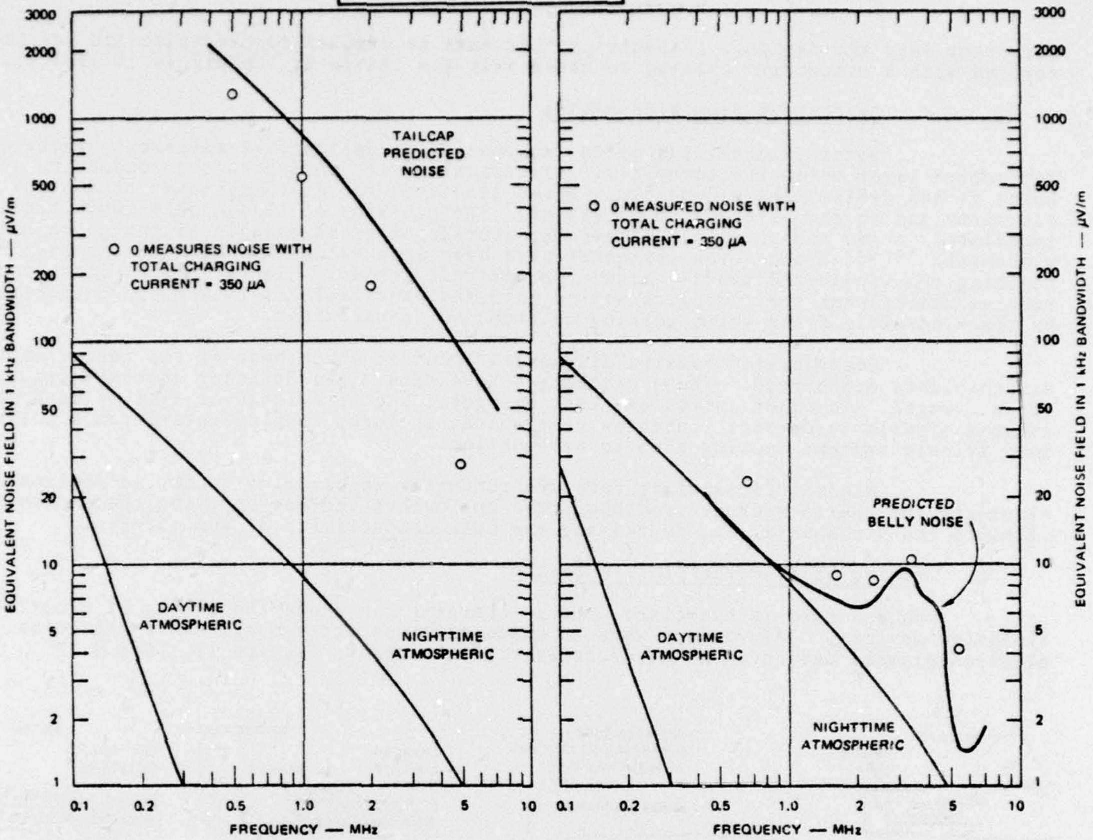

FIGURE 13 CORONA DISCHARGE NOISE IN 707 ANTENNAS

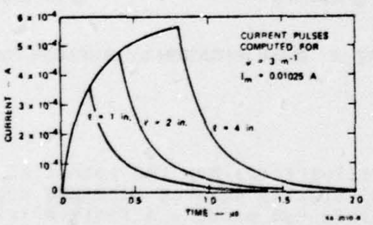

FIGURE 14 TYPICAL CURRENT PULSES INDUCED BY STREAMER DISCHARGES

Assuming that the wire is terminated in an impedance of 1000 ohms the current pulses of Figure 14 will generate voltage pulses of 0.4 V. This pulse amplitude is sufficient to upset a variety of circuits.

The charge transferred by a single streamer discharge is 1 to 1.5×10^{-9} Coul. This is roughly the same as the charge transfer in a corona pulse. The difference in pulseforms produced by the two mechanisms results from the difference in the lengths of the two discharges. The corona discharge extends to only one tip radius from the burr or other imperfection from which it occurs. A streamer on the other hand extends many inches out on to the dielectric. This long discharge length causes the streamer to contain substantial low frequency energy.

Recently there is increasing evidence that streamer discharges generate more high-frequency energy than was evident in early studies. Further work to better characterize the streamer breakdown could be very profitable at this time.

2. Effects on Systems

Many of the practical problems associated with surface streamer discharges have had to do with dielectric antenna covers or with insulators forming part of the antenna. For example, a 1-inch-diameter piece of plastic protruding forward from a towel rail antenna on a commercial aircraft caused the ADF system to malfunction every time the aircraft flew in precipitation. Antennas located under pilot's canopies are highly susceptible to noise generated by streamers on the exterior of the canopy.[17,18]

Great care should be exercised not to route digital system signal cables under dielectrics located on aircraft frontal surfaces. The amplitude and waveform of the induced pulses is such that they can readily cause system upset.

3. Techniques for Elimination

Streamer discharge problems can be eliminated by locating dielectric surfaces so that they are not struck by precipitation particles. This can be achieved by placing the dielectric materials far enough aft on the aircraft that they are not exposed to particles (see Figure 4). Alternatively, it is possible to incorporate a metal particle

deflector into the design. Dielectrics that must be exposed to precipitation can be covered with a conductive coating to bleed away the charge as rapidly as it arrives.

4. Discharges From Windshields

Modern windshields often incorporate an optically-transparent electrically-conductive layer below the outer ply. Electrical power applied to the conductive layer heats it and de-ices the windshield. Wires lead from the de-icing layer to control circuitry and to the aircraft power system. The presence of this highly-conductive layer immediately below the outer ply allows the storage of great amounts of charge on the windshield.[40-43] Ground crew personnel have been shocked and knocked off aircraft by touching the windshield shortly after the aircraft landed. Fighter pilots occasionally receive shocks when their fingers extend onto the windshield as they support themselves by the windshield frame while getting up from the forward seat.

Occasionally massive discharges occur in which most of the charge on the windshield is discharged. These discharges have caused the de-icing control circuitry to be damaged. In other instances, less energetic but more frequent discharges have induced signals in de-icer control wiring which, in turn, coupled interference pulses into avionic systems causing them to malfunction.

Windshield manufacturers are currently endeavoring to devise designs to eliminate the charge storage problems. The approaches include applying conductive surface films to the windshield, and increasing the bulk conductivity of the outer ply.

C. Firing of Electroexplosive Devices

Unless care is exercised, static charging can cause the firing of electro-explosive devices. The various ways in which electrostatic energy can initiate an electroexplosive device on an aircraft are illustrated in Figures 15, 16 and 17.

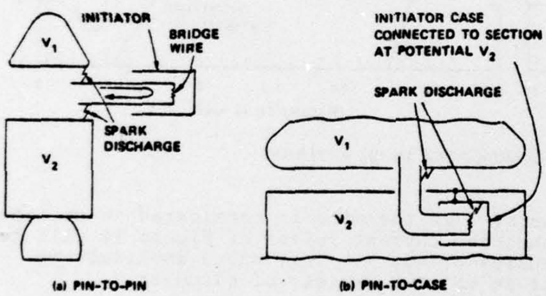

FIGURE 15 SQUIB INITIATION BY SPARKS BETWEEN SECTIONS

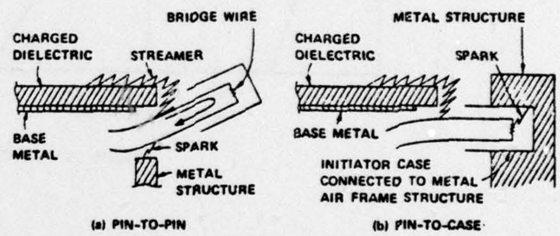

FIGURE 16 SQUIB INITIATION BY STREAMER DISCHARGE

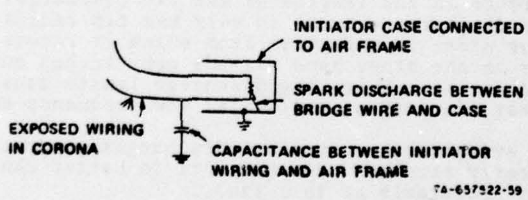

FIGURE 17 SQUIB INITIATION BY CORONA DISCHARGE

Figure 15 illustrates how the potential difference existing between unbonded sections of the vehicle can produce a spark which may initiate a squib in the pin-to-pin or pin-to-case mode. The ways in which streamer discharges from a plastic surface might initiate a squib are illustrated in Figure 16. If squib wiring is exposed to high electric fields, as indicated in Figure 17, corona discharges occurring from the wiring can cause a spark between bridge-wire and case to initiate the electro-explosive device.

Fortunately, the design of electroexplosive devices is improving. Modern units are sufficiently insensitive in both the pin-to-pin mode and in the pin-to-case mode that they are difficult to initiate with the amount of static charge normally available on aircraft.

D. Dielectric Deterioration

Finally, static charging of dielectric surfaces can result in the deterioration of the dielectric. For example, if a large radome is placed over a radar dish on the nose of the aircraft, the outer surface of the radome will become charged. The charge can return to the airframe either by streamers over the outside surface or by penetrating the radome and sparking to the radar dish. When the radome is punctured, moisture collects in the holes and causes delamination of the material and, eventually complete deterioration of the radome.

Similarly, there have been reports of damage to the outer glass plys of windshields. On some modern aircraft, the windshield is so large that charge is relieved by

puncturing the outer ply rather than by a surface discharge along the outer surface. The outer ply must therefore be made sufficiently thick to withstand the voltage needed to sustain a surface discharge.

E. Precautionary Measures

Insofar as static electricity effects are concerned, the ideal aerospace vehicle would be a completely bonded metal structure equipped with noise reducing dischargers at appropriate locations. Careful bonding of the adjacent metal parts prevents sparking from section to section. Since the vehicle is made completely of metal, streamer discharges cannot occur. Thus, the only electrical breakdown possible is corona discharge from the extremeties, and the use of noise-reducing dischargers eliminates any problems associated with corona.

Such a structure, of course, is not possible for various reasons. First, to see out of the vehicle and to communicate from it requires an antenna and a windshield; these use dielectric materials, some of which must be located on frontal surfaces where they can become charged. Metal corrosion is forcing direct metal-to-metal contact at structural joints to be reexamined, which may lead to poor electrical bonding of the structure. Finally, weight considerations are forcing more and more of the aircraft to be fabricated of plastic. Practical aircraft, therefore, will occasionally have imperfect bonds and plastic surfaces will be included in the structure. Insofar as static electricity is concerned, plastic surfaces cause no trouble provided they are not exposed to impinging precipitation particles. Thus, plastic for trailing edges and perhaps for body sections will probably generate very little radio frequency noise, although it will complicate the installation of corona dischargers and aggravates the lightning protection problem. If plastic materials must be located on frontal surfaces of aircraft, they should be shielded from impinging precipitation particles by a particle deflector, the outer surface of the plastic should be made conductive to drain away the charge, or the entire plastic piece should be conductive. Such a treatment will prevent discharges (thereby eliminating noise) and will prevent charge build-up and puncture which damages the dielectric itself.

IV SUMMARY

In summary, we have observed various forms of static electrification that are possible on airborne vehicles. The charging processes cannot be eliminated, but their deleterious effects can be minimized through proper vehicle structural and systems design. Design changes that are likely to modify the electrical bonding of aircraft structure should be carefully reviewed to make certain that they do not cause problems to the system. When plastic materials are to be used on the vehicle, use of the material should be examined to ensure that it will not cause problems in any of the pyrotechnic or communication and navigational systems. If plastic material is used on a frontal surface, it should be treated to ensure that static electrical charges do not cause rapid deterioration. The noise generated by corona discharge from the aircraft can be minimized through the proper installation of an adequate number of modern aircraft dischargers.

Most of the decisions regarding the structural form and the materials used in fabricating an aircraft, of course, are not made by people well versed in static electricity and its effects. It is hoped that this brief review will provide some insight into the problems that can result from relatively trivial changes in structure if static electrical effects are not borne in mind.

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QUESTIONS and ANSWERS

1 – F G. Odam

Q – Should a radome antistatic treatment be applied over or under the rain erosion coat ?

A – Ideally, the antistatic coating should be the outermost layer on the radome. Unfortunately, imparting conductivity to the coating generally degrades its mechanical properties sufficiently that it is not able to withstand rain and ice crystal impact. For this reason, the abrasion resistent erosion coat is applied over the antistatic treatment in most systems. The argument is that the erosion coat is a sufficiently poor dielectric that it will be punctured by charge accumulating on its outer surface substantially before surface flashover occurs.

2 – From S.D. Schneider

Q – Did you verify your estimate or 1000 pF for the helicopter capacitance ?

A – The capacitance of 1000 pF is based on measurements. See pL35 of "Lightning and Static Electricity Conference Papers" 12-15 December 1972, Las Vegas, Nevada. These date, presented by Born & Durbin, agree with our data generated is laboratory scale model measurements of capacitance.

3 – From R.H. Evans

Q – At one time there was considerable research carried out on the possible use of active electrostatic dischargers, that is dischargers connected to a high voltage power supply. Is there still an interest in this type of discharger, particularly for helicopters ?

A – Active dischargers work well on conventional aircraft, and it was felt that the technology could be transferred to helicopters. Unfortunately, it was found in the work of Ref. 26 & 35 that the dust cloud stirred up by a hovering helicopter was so highly charged that the fields generated by the cloud dominated the region about the helicopter. Thus, to measure helicopter-to-ground potential, one could not rely on the readings of one or more field meters strategically located on the helicopter. Instead, it would be necessary to touch the ground with one terminal of the sensing system. If it is necessary to establish ground contact to obtain a reliable reading of helicopter-to-ground potential, it was felt that it would be simpler to use passive discharging systems as discussed in Ref. 23 for dusty environments.

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FUEL ELECTRIFICATION

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SUMMARY

A brief review is given of the principal physical processes involved in charge generation and dissipation when low conductivity fuels flow in pipes. Hazards which arise when charged fuel flows into metal or plastic tanks are described.

With increased use of plastics and low conductivity composite materials in aircraft research into the potential electrostatic hazards involved has been carried out in a number of centres. Some of the new techniques being involved are discussed in the paper. They include the use of charge collecting electrodes in plastic tanks and conducting polymers, to complement the use of anti-static additives in the fuel. An active discharger system which can deliver electrically neutral fuel to the tank is described.

1. INTRODUCTION

Until recently, the annual number of aircraft fuelling fires caused by electrostatic phenomena has been negligible. The incentive to make fundamental studies of charging phenomena was therefore small, in view of the limited scientific effort available. Recently, however, low conductivity composite materials and plastics have been increasingly used for fuel tanks. Such materials have been used for road and rail tank trucks and some military aircraft have been built with fibre reinforced plastic fuel tanks.

In the U.K. the Ministry of Defence has begun a ship construction programme for mine counter-measure vessels constructed entirely of glass reinforced plastic (G.R.P). A prototype ship, H.M.S. Wilton, has been in service with the Royal Navy since 1973 and the first of a class of larger vessels, H.M.S. Brecon, was launched in July 1978. A research project was set up at Southampton in collaboration with the National Gas Turbine Establishment (Cobham) in 1975 as part of a support programme for the G.R.P. ship project. A large fuel test facility was set up at Northam, Southampton in 1974, having two 2,500 gallon tanks in G.R.P. and aluminium(1). This facility is currently used for pure and applied electrostatics research on behalf of the U.K. Ministry of Defence and the U.S.A.F. New instruments and discharger systems are being developed and the rig can be used to simulate potentially hazardous fuelling situations under controlled conditions.

In the United States at least seven minor fires or low-level explosions have occurred since 1975 during fuelling of military aircraft with JP4. The incidents were associated with the incorporation of polyurethane foam as an explosion suppressor. Some of the charging phenomena occurring in the tanks were similar to those encountered in the M.C.M.V. work.

It is not appropriate in this paper to give a comprehensive review of all electrostatic fuel hazards work. An excellent basic text-book dealing with fundamentals is that by Klinkenburg and van der Minne(2). Good discussions of the problem of safety criteria in metal tanks are contained in papers by Strawson(3) and Krämer, Asano and Schön(4). A recent review by Bright(5) attempts to bridge the gap between the early work on metal tanks and more recent work with composite materials.

A useful recent report on aircraft fuelling at high rates by Hörnberg et al(6) discusses aircraft fuelling operations.

In the present paper some of the results of current work at Southampton will be reported together with a discussion of some of the methods being developed to reduce hazard levels during fuelling.

2. FUNDAMENTAL PROCESSES

When a stationary dielectric liquid is in contact with a metal surface, a complex layer of electric charge is established at the liquid/metal interface, known as the 'double-layer'(7). The charge layer adjacent to the metal is strongly bound, while a layer of roughly equal and opposite charge separated from this bound layer by the double-layer thickness δ is mobile and free to move with the liquid (see Fig. 1). A potential difference is established across the double layer and charge transfer can occur across it. (ζ) Charges can flow from the metal into the liquid and also charge exchange can occur as liquid neutral molecules or ions make contact with the metal wall. If bulk motion occurs, the mobile layer of charges constitutes a current i_s, known as the streaming current. For a pipe of diameter d we see that

$$i_s = \frac{\pi d^2}{4} \rho \nu \qquad (1)$$

(ρ = charge density ν = flow velocity)

For laminar flow, Klinkenberg(2) showed that

$$i_s = \pi d \epsilon_o \epsilon_r \zeta \nu/\delta \qquad (2)$$

However, in most practical fuelling cases one has turbulent flow. In this case it is difficult to compute i_s. For pipes of infinite length Klinkenberg derived an expression:

$$i_s = \pi d \tau j_a \nu \left| 1 - \exp {}^{e}/_{\nu\tau} \right| \tag{3}$$

τ = relaxation time, ν = flow velocity, j_a = current density from pipe wall.

Unfortunately, j_a cannot be calculated. Bright attempted to calculate the charge density ρ from first principles, obtaining an expression(5)

$$\rho = \frac{4\, \varepsilon_o\, \varepsilon_r\, \delta}{\mu\, R\, \tau} = \frac{4\, \varepsilon_o\, \varepsilon_r}{\mu\, R} \left(\frac{\Delta_m}{\tau}\right)^{\frac{1}{2}} \tag{4}$$

Δ_m = molecular diffusion coefficient, μ= ion mobility, R = pipe radius.

$$\text{Hence,} \quad i_s = \frac{\pi\, \varepsilon_o\, \varepsilon_r\, d^2\, \nu}{\mu} \left(\frac{\Delta_m}{\tau}\right)^{\frac{1}{2}} \tag{5}$$

Eq(4) predicts the general observed behaviour of ρ for hydrocarbon fuels, where it is found in general that ρ is independent of ν. However, Eq(4) cannot be used to predict charge densities in real fuels because of the very considerable influence of trace impurities on the double layer and hence on charge generation. Eq(4) was derived using a very simplified model which ignores the influence of adsorbed ions.

Fig. 2 shows a series of results by Klinkenburg in which ρ was measured as a function of flow rate using two samples of kerosine - one containing a corrosion inhibitor and the other an anti-static additive. The pressure of a few p.p.m. of additive increased fuel conductivity from 0.81 p s/m for undoped fuel to values from 4.1 to 28 p s/m. The corresponding range of charge density was from 6 $\mu c/m^3$ to 300 $\mu c/m^3$. Conductivities in this range are found in aircraft fuels and similar (unpredictable) variations in charge density would be expected.

Most of the charge generation in a typical aircraft fuelling installation occurs in the filter-separator. This is because of the large surface area in contact with fuel and also because interfacial phenomena associated with H_2O can give rise to enhanced charge generation as discussed above. There is at present no available theoretical model to predict charge generation in filters. Leonard(8) has carried out extensive studies of charging in filter elements and porous materials. He found that no simple relationship existed between conductivity and charging tendency. Low levels (p.p.m.) of H_2O in conjunction with trace impurities had a marked effect on charging tendency, sometimes with little or no effect on conductivity.

3. SAFETY CRITERIA IN FUEL TANKS

A hazardous situation will occur in a tank when sufficient charge has accumulated to give rise to a sufficiently high electric field to result in a spark or brush discharge. Ignition will occur if the discharges occur in an inflammable atmosphere and if ignition energy requirements are satisfied.

The two critical periods are in general:-

1. During initial stages before the fill pipe outlet is submerged;
2. As the fuel surface approaches the top cover of the tank.

With (1), limitation of initial fuel velocity to 1 m/sec will in general remove the hazard. With (2) there is disagreement about the most useful criterion to use. In the author's opinion, fuel surface potential V_s appears to be the most satisfactory. Strawson and Lyle support this view(9). Their experiments suggest that a value for V_s of about 45 kV in the absence of an earthed probe represents the upper limit for safety. Work at Southampton(10) has indicated that V_s must not exceed 20 kV in the presence of an earthed probe. Fuel tanks generally have complex geometries with metal fittings or protrusions extending towards the fuel surface. With V_s < 20 kV it has not been possible in the Southampton work to produce ignitions in propane-air mixtures.

3.1 Metal Tanks

Obviously the potential distribution in the tank will principally be determined by the charge acquired by the tank. This will be a function of i_s, charge density, the relaxation time of the fuel τ and the dimensions of the tank. (Note $\tau = \varepsilon_o\, \varepsilon_r\, \rho$ where ρ is fuel resistivity)

After time t , the charge density ρ_t in the tank will be:-

$$\rho_t = \frac{\rho\, W\, \tau}{V} \left| 1 - \exp \frac{-t}{\tau} \right| \tag{6}$$

where W = volume flow rate, v = tank volume, ρ = inlet charge density

Strawson(3) has obtained experimental relationships which permit one to relate volume flow rate V to the surface potential V_s. For a permitted maximum V_s one can determine V for a given tank size.

The parameter $\left|\frac{\rho}{\nu\delta}\right|$ is found to be critical (ν = flow velocity, δ = fuel conductivity).

A value of 12.5 is currently regarded as the limiting safe value for a range of δ from 0.4 - 8 p S/m. For conductivities > 8 p S/m Southampton work suggests that ρ is independent of velocity and the Strawson approach, which assumes $\rho \propto \nu$, may require modification.

In general, Eq(6) demonstrates that ρ_t and hence the hazard in the tank can be reduced by increasing relaxation time τ. Almost all the safety measures currently in use involve this approach. Recent developments at Southampton have led to a method of reducing inlet charge density ρ to zero.

3.2 Safety criteria in plastic tanks

With tanks having high resistivity walls, charge leakage to ground cannot readily occur. Charge tends to migrate to the walls where it is stored, the walls constituting the dielectric of a capacitor, Fig. 3a. Charge decay rates are independent of fuel conductivity and depend on the bulk and surface conductivity of the tank.

The total charge Q_t remaining in the tank after time t is given by:-

$$Q_t = i_s t \left[1 - \exp \frac{-t_f}{\tau_t}\right] \tag{7}$$

τ_t = relaxation time of absorbed charge in tank, t = fill time.

Typical values of τ_t for G.R.P. are 10-30 minutes.

$$i_s = \pi r^2 \rho \nu \qquad \therefore Q_t = r^2 \rho \nu t \tag{8}$$

Treating the base of the tank as an earthed capacitor, one can easily show that the potential developed across the base of the tank V_b is given by

$$V_b = \frac{i_s t_f d}{\varepsilon_0 \varepsilon_r a^2} \tag{9}$$

(assuming a square tank of side a, wall thickness d)

With practical values of i_s measurements at Southampton have shown that potentials of > 300 kV can be established in the base of the tank. Up to 90% of the charge in the tank is stored in this way.

The fuel surface potential is related to V_b in a complex way and its determination will be discussed later.

Plastic tanks present two principal electrostatic hazards.

1. Possible surface discharges when permissibile V_s exceeded.
2. Possibility of energy storage in tank walls leading to:-

   2.1 Discharge to surface from base;

   2.2 Puncturing of tank wall.

The incorporation of reticulated plaster foam in metal tanks as a flame inhibitor can give rise to electrostatic hazards somewhat related to the above discussion.

Some work at Southampton on plastic tanks and foam filled [illegible] al tanks will be described. This work was concerned with specific ship and aircraft projects but has ge[illegible]al applications.

4. G.R.P. TANK STUDIES

The aim of this work was to investigate hazards associated with fuelling of mine counter-measure vessels (M.C.M.V.) of G.R.P. construction. A large test facility was set up in 1974 including a 2,500 gallon G.R.P. tank (2m x 2m x 2m). A schematic diagram of the rig is shown in Fig. 3. A novel feature of this installation is the use of charge injection to control the charge input level to the tank. This is described in detail in Ref. 9. This system uses a high voltage diode injector to introduce a controlled streaming current i_s* into the tank. Also a novel charge density meter(10) having no moving parts is used to determine ρ. Diesel fuel in a conductivity range (10-30 p S/m) was used as the test fuel.

Experiments with the rig have been carried out in conjunction with N.G.T.E. (Cobham) and the Navy Department and some results have been published in Ref. 11,12. A typical result is shown in Fig. 4. Although τ for the fuel was 2 secs, the relaxation time of the stored charge was ≈ 600 secs. The increase of electric field after filling is due to charge migration to the wall.

Ignition experiments by Haig(12) showed that ignition of propane-air mixtures could occur when the surface potential of the fuel exceeded 25 kV, due to fuel-metal electrode discharges. Experiments on the large rig were carried out and it was found that similar discharges were noted. Below approximately 20 kV conical protruberances (Taylor cones) pulled up from the fuel surface by the electric field bridged the gap and no discharge occurred.

The surface potential V_S was measured as a function of input charge density ρ at 300 g.p.m. (see Fig. 5). From these experiments a value of ρ of 125 c/m³ was established as the maximum permissible

charge density. Below this value it would be impossible to produce a discharge of sufficient energy to produce ignition.

On two occasions, when filling at 300 g.p.m. at ρ = 150 $\mu c/m^3$ very high energy discharges were noted (10-50 J) audible at a range of 50 m. These were apparently associated with charge storage in the base of the tank.

The potential V_b at the base of the tank was determined by measuring the current flowing through the base to electrodes of known area (10^6 cm^2 and 10^4 cm^2). The volume resistivity of the tank was 1.36 . 10^{13} Ωm. With ρ = 5 $\mu c/m^3$, V_b = 12.6 kV. For ρ = 100 c/m^3, V_b = 300 kV. This very high potential in the tank had clearly been responsible for the two high energy discharges mentioned above.

A computer study was set up in collaboration with the Rutherford Laboratory and a finite distribution in a plastic tank, taking into account the non-uniform distribution of charge. The model which evolved has given excellent agreement with measurements and has enabled discharge systems to be devised for the M.C.M.V. ship H.M.S. Brecon(12). Examples of results are given in Figs. 6, 7. Fig. 6 shows the potential distribution in the tank after filling at 300 g.p.m. for ρ = 100 $\mu c/m^3$. An earthed probe (a field meter) is assumed to be suspended above the tank. The high stress concentration at the base of the tank is clearly seen. Fig. 7 shows how the potential at the base of the tank can greatly exceed the surface potential.

This study is being extended to include 3D presentation of the potential distribution in tanks of complex geometry in collaboration with Professor J. R. Smith (Aberdeen University).

5. DISCHARGE METHODS

Two methods for elimination of static hazards have been developed and tested at Southampton.

1. Active dischargers;
2. Earth plates.

5.1 Active dischargers

Here a high voltage liquid diode is used to inject charge into the fuel(11). By measuring the charge density in the fuel and incorporating this sensor and the discharger in a feedback loop one can fill the tank at any rate maintaining zero charge density in the tank.

This method has been successfully used at charge density levels of over 100 $\mu c/m^3$. Response time is rapid, as shown in Fig. 8. The system is obviously suitable for incorporation in aircraft fuelling systems and a commercial prototype will shortly be evaluated using JP4 by the U.S.A.F.

5.2 Earthing Plates

The field distribution shown in Fig. 6 suggests that a conducting plate at ground potential should act as an efficient discharger. Experiments showed that with earthed stainless steel plates or steel meshes 1.0 m x 0.5 m the electric field due to the fuel could be effectively neutralised even when filling the 2,500 gallon tank at 250 g.p.m. with ρ = 185 $\mu c/m^3$. All the input streaming current i_s = 2.3 μA was collected by the electrode.

This simple and effective measure can only be used when the base of the tank is grounded (or is immersed in the sea).

6. FUEL SLOSHING

During sea trials with H.M.S. Wilton, it was found that high speed 360^o turns of the ship produced 'sloshing' of the fuel in the tank with 50% filled tanks. This movement of fuel relative to the tank walls resulted in the production of very high charge densities in the fuel. The trials are to be repeated

This phenomenon has not previously been reported and clearly requires further investigation.

7. AIRCRAFT FUEL TANK STUDIES

Many U.S.A.F. helicopters and fixed wing aircraft have their fuel tanks filled with polyester or polyurethane foam for explosion suppression. Following a series of minor fires and low-level explosions during fuelling such tanks, the U.S.A.F. has set up an extensive investigation. These foams are very successful as explosion suppressors and if the associated electrostatic problem can be eliminated their use could be contemplated in other applications. Work at Southampton has mainly been directed to:-

1. Development of conducting foams;
2. Development of active dischargers for aircraft fuelling.

When fuel is pumped into a polymer foam, charge exchange occurs as in with the filter-separator. If the fuel inlet jet forces back the foam to produce a vapour space, a potential can be developed between nozzle and foam and ignitions can occur. Leonard(13) and Farrer at Southampton have measured charging levels in the standard U.S.A.F. foams and found that high levels of charge density can be produced.

One standard method of reducing the electrostatic hazard is to use anti-static fuel additives such as ASA3. Leonard recommends that with ASA3 a fuel conductivity well above 100 p s/m would be required in order to avoid charge generation problems with plastic foam.

7.1 Work on conducting foams

The approach at Southampton has been to attempt to develop conducting foams which can dissipate charge. The advantage of this is that it would not be necessary to use anti-static additives. The effectiveness of additives such as A.S.A. 3 should be increased with conducting foam as the fuel/polymer surface area would be greatly increased.

The aim of the work was to develop polyether foam having a resistivity $< 10^9\ \Omega m$ as work in other fields suggested that this would provide rapid discharge in the fuel. The main characteristics of the foam presenting most problems in the U.S.A.F. application was:- $\rho = 3 \cdot 10^{13}\ \Omega m$ .8-16 cells/inch. A first target was to produce a foam having $\rho = 10^9\ \Omega m$. Working in conjunction with ICI Organics Division several samples of foam were produced loaded with carbon black, carbon fibre (~ 3 mm length) and a conducting salt. In all 16 samples were produced, the most promising being a combination of 1% fibre and 5% salt, when a value for ρ of $1 \cdot 10^9\ \Omega m$ was obtained. The pore size for these foams was 100 cells/inch.

An alternative approach was to take the standard U.S.A.F. foam and to coat it with a conducting polymer. This work was undertaken in conjunction with Canespa Ltd. Using a carbon loaded vinyl coating a foam having $\rho = 10^5\ \Omega m$ was produced. Further samples having $\rho = 3 \cdot 10^5\ \Omega m$ were produced. These were immersed in JP4 for 5 weeks without charge in fuel conductivity, suggesting that the coating is stable. The increase in foam conductivity of 10^8 times was encouraging.

Using a rig similar to the Exxon Mini Static Tester, charge generation due to foam samples 3" x 2½" diameter was measured. Results are shown in Fig. 9. This shows a linear relationship between $(\rho)^{\frac{1}{2}}$ and the cells/inch of the foam, suggesting that charge generation is α foam area. This result also is consistent with high charge densities produced in fuel filler elements. Lowest charge density was obtained with foam having a pore size in the 8-17/inch range. It was therefore decided to produce samples of standard coated U.S.A.F. foam for testing in a fuel tank with JP4.

The experiments are in an early stage, but preliminary results with a small test rig are encouraging. The arrangement is shown in Fig. 10. Fuel was injection charged to levels of about 2000 $\mu c/m^3$. Foam having a resistivity of $3 \cdot 10^5\ \Omega m$ was used. The field at the surface of the tank and the collected current were measured with and without foam.

TABLE 1

Test with conducting foam ($\rho = 3 \cdot 10^5\ \Omega m$)

Tank Filling Tank Filling	Injector (kV)	Indicated Max. Field (V/m)	Surface potential (V)	Collected current (A)	Charge density ($\mu c/m^3$)
JP4	-10	29500	1475	$-1.7 \cdot 10^{-7}$	1930
JP4 + conducting foam	-10	270	13	$-1.5 \cdot 10^{-7}$	1670
JP4 + USAF Foam	-10	30200	1510	$-1.25 \cdot 10^{-7}$	1390

Table 1 shows that the observed surface potential was reduced by a factor of over 100 with conducting foam to only 13 V. Fig. 10 shows the build-up and decay of field in the tanks and clearly illustrates the effect of conducting foam.

Experiments will take place shortly with a more complex rig having larger tanks. Conducting foams and also expanded metal will be tested as well as new types of nozzle. Hopefully the conducting foams being developed will prevent hazardous charge levels occurring during aircraft fuelling even with undoped (low conductivity) JP4.

8. CONCLUSIONS

Filter elements are the principal charging source in fuelling systems. Where non-conducting materials are used in fuel tanks there is generally an increased hazard level. Fuel sloshing can produce high charge levels in plastic tanks.

In addition to charge dissipators such as ASA3, the following new approaches show promise:-

1. Monitoring of input charge density to tank.
2. Neutralisation of charge in tanks by active discharger.
3. Use of small area earth plates in low conductivity tanks.
4. Use of high conductivity polyether foams in tanks.

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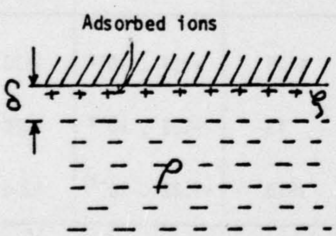
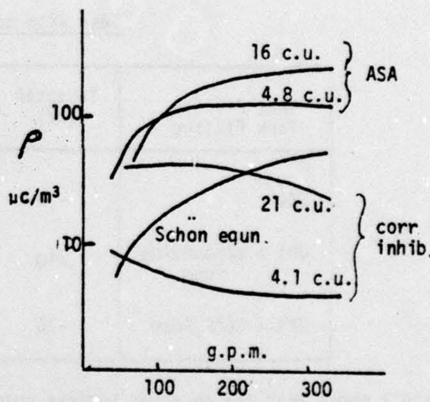


Figure 1



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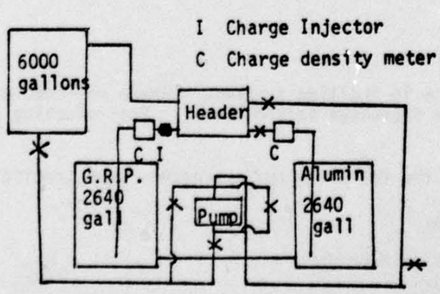

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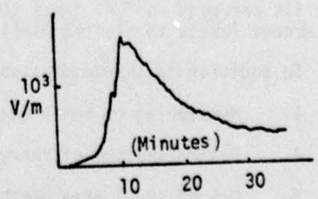

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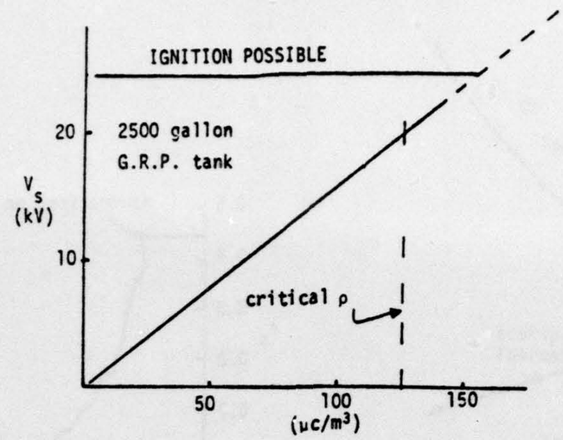

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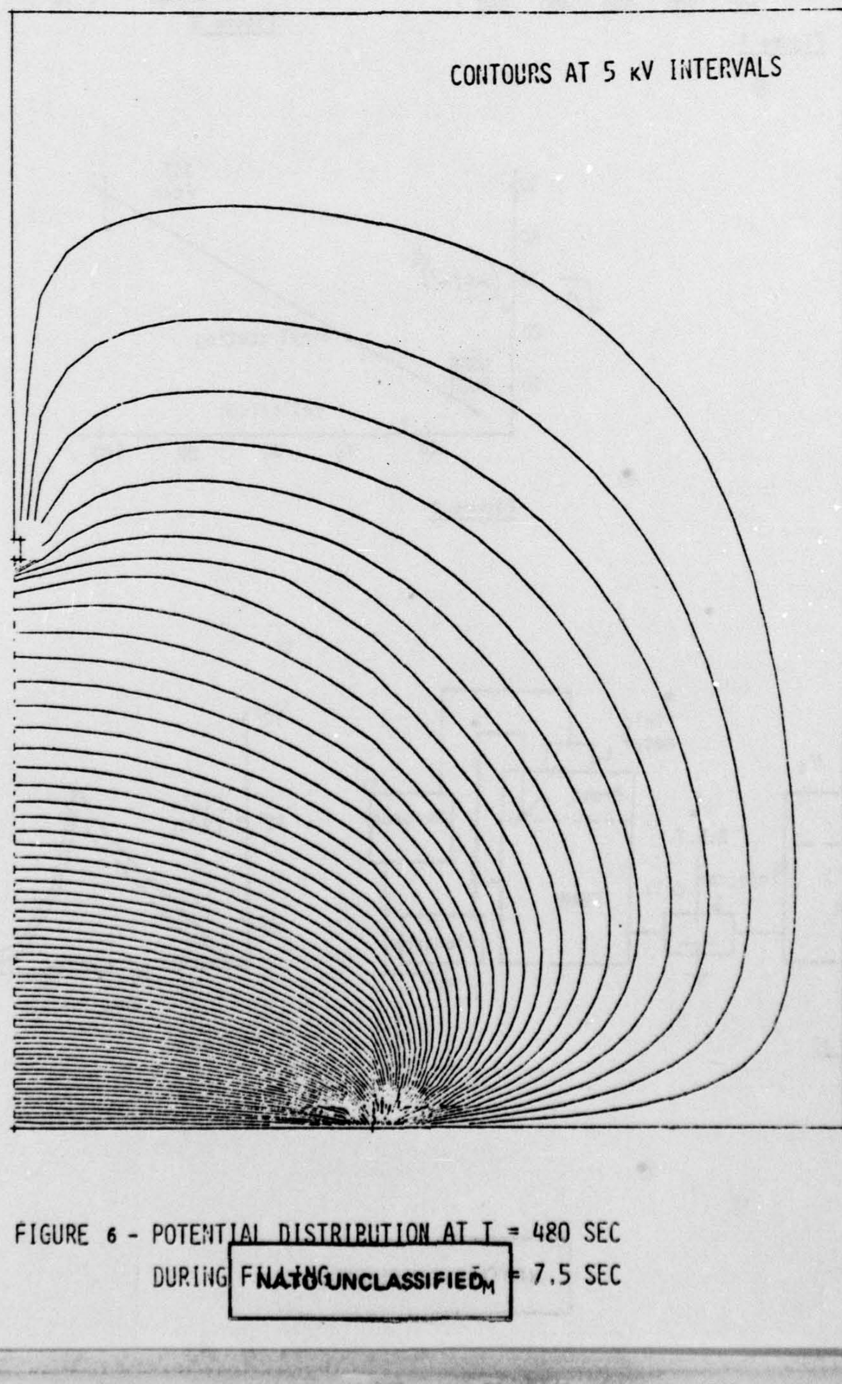

FIGURE 6 - POTENTIAL DISTRIBUTION AT T = 480 SEC
DURING FILLING τ_M = 7.5 SEC

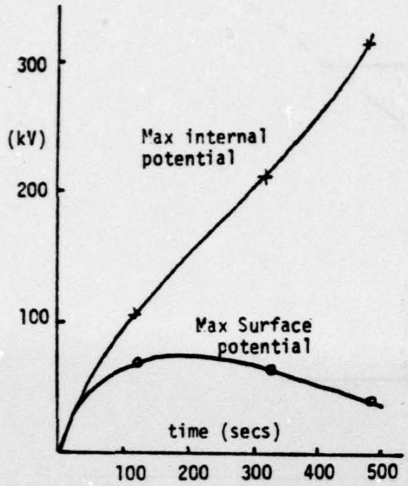

Figure 7

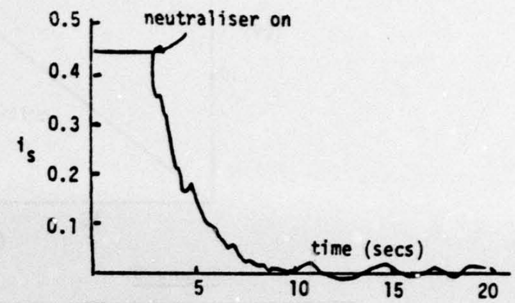

Figure 8

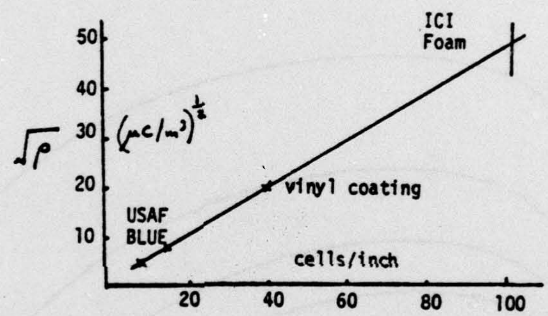

Figure 9

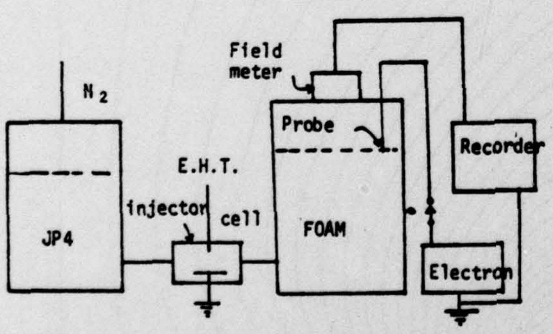

Figure 10

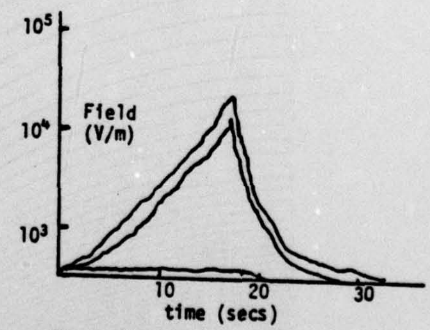

Figure 11

BIOGRAPHICAL NOTE

A.W. BRIGHT

Alfred William Bright received his BS degree in Electrical Engineering from Queen Mary College in 1944. After a two year stint as development engineer at Marconi Wireless Telegraph Company, he returned to Queen Mary College and received his Ph.D in EE in 1949. Subsequently he did post doctoral work as Attaché de Recherche at Grenoble University and was Scientific Officer. Services Electronics Research Laboratory of the Royal Naval Scientific Service from 1951-52. He served as Lecturer in Applied Electron Physics, Imperial College of Science and Technology from 1953 to 1960 and served as Managing Director S.A.M.E.S. (Great Britain) Limited from 1960 to 1965. He has been at Southampton University since 1965, most recently serving as Professor of Applied Electrostatics and Director of the Applied Electrostatics Group. Dr. Bright has been a Fellow of the Institute of Physics since 1964 and was appointed Chevalier de l'Ordre National du Mérite by the French Government in 1975. His present research interests are in electrostatic hazards in industry and electrostatic powder coating techniques.

AIRCRAFT STATIC CHARGING TESTING

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Summary

Aircraft protection against static charge accumulation involves minimum-noise electrostatic dischargers, perfect bonding of all conductors, antistatic painting of radomes and antenna covers, and conductive coatings of canopies. This paper deals with the procedure proposed by ONERA for :

a) characterizing in the laboratory the effects of aircraft charging on the operation of unprotected navigation and communication subsystems ; b) assessing in the laboratory the validity of the above mentioned methods of protection ; c) checking in the factory the correct application of these methods ; d) testing routinely between flights, in the field, the good condition of the protecting devices and treated surfaces.

It is hoped that careful application of this procedure will, in the near future, increase the reliability of navigation/communication subsystems during their operational use.

I. INTRODUCTION

Static charge accumulation on aircraft structures can bring about three types of discharges : sparks between conductors, streamers over the insulating surfaces, and coronas at the sharp points and edges. All these discharges induce radioelectric noise on navigation/communication equipment, reducing its operational performance and impairing flight safety.

The principles of a good protection against these harmful effects are well known [1]. Sparks can be avoided by a careful bonding of all the metallic parts of the aircraft. Superficial electric charges developed by friction or collected on insulators can be evacuated before initiation of a flashover, if the surface resistance is kept below a certain value by using conductive *paints or coatings* on windshields, canopies, radomes and fairings. Lastly, the coupling between corona discharges and antennas of the communication/navigation sub-systems can be reduced drastically by *a careful choice* of the configuration, the position and the voltage threshold of the passive dischargers used for draining the static charges from the aircraft.

However, the practical application of these principles is today a matter of empiricism. On the one hand, apart from the possibility of performing a full in-flight experiment [2] [3], there is not yet a standard method for characterizing on a real aircraft, treated or untreated, all the disturbances induced in navigation/communication subsystems by the accumulation of charges. On the second hand, the in-flight experiment is an expensive task : the complexity of the full avionic system with its many interfering subsystems, the non reproducibility of the meteorologic situation from flight to flight, the inconvenience of recording, storing, packing and correlating a huge quantity of data might turn such an experiment into a heavy burden for scanty results if these experiments have not been prepared by previous tests on the ground.

The situation is still worse if we go from the laboratory to the factory or to the airfield. No tool exists to verify, at the production stage, if the protection methods have been correctly applied during the construction of the aircraft ; no standard testing procedure has been elaborated for trial by the buyer. At the exploitation stage, no airline is in a position to carry out maintenance tests routinely between flights for verifying the good condition of the protective devices and of the treated surfaces. Meanwhile, from the field engineers to the pilots, everybody is convinced that such a possibility of verification is badly needed : the dangerous disturbance of vital subsystems induced by the discharges of static electricity is an universal worry.

The present work constitutes a step towards the definition of a standard procedure for testing aircraft charging phenomena and protections, in the laboratory, in the factory and in the field. Until such a procedure is operational and universally adopted by the research institutions, the aerospace industries and the airlines, full protection against communication/navigation disturbances is not ensured during flight in very adverse atmospheric conditions.

In this paper, the solutions proposed by ONERA to meet the requirements of a safe protection are outlined. In order to apply the test method quickly and efficiently, specific instruments have been designed and implemented at ONERA, and will be available on the market in the near future. A general description of their performance is given below. Lastly, a practical application is also described in some detail ; in cooperation with the Company "Avions Marcel Dassault-Bréguet Aviation", the proposed procedure has been applied to the Falcon 10 business aircraft and to the French-German training aircraft, the Alpha-jet, opening the way for decisive progress in the protection of these aircraft against disturbances due to electric charging. The methodology used during one of these tests and some of the most representative results will be presented.

II. TEST PROCEDURE REQUIREMENTS

The proposed test method shall fulfil the following functions :

a) check the bonding between metallic surfaces, even if these surfaces are covered by an insulating coating ;

b) measure the value of surface resistance of semi-conductive coatings or resistive paints deposited over insulating substrate, even if these coatings are covered by a layer of higly insulating material ;

c) simulate tribo-electric charging, or charge collection, separately on any surface element of the aircraft, in order to determine the location and the nature of the more sensitive spots, and to assess the effect of the charging of these elements on the navigation/communication subsystems ;

d) verify that the coupling between dischargers and antennas is minimum, i.e. that the position of the dischargers is optimized, and that the noise collected by the antennas for a given discharge current is in conformity with the value announced in the specifications and has not been increased by some deterioration experienced by the dischargers.

During all this testing procedure, the aircraft shall be grounded, or, if global operation of the dischargers is required, it shall be insulated by dielectric slabs placed below the wheels. This insulation shall be matched with the voltage limit due to the operation of the dischargers, in order to avoid flashover between aircraft and ground. A high resistance bleeder shall be used, for safety, to discharge the aircraft below the threshold voltage of the dischargers.

If any voltage generator is used for operations (c) or (d), its output impedance shall be kept very high, to insure safe operation.

Operations (a) and (b) are to be performed without damaging the superficial layer of paint.

Operation (c) shall be clean, i.e. without projection of particles or aerosols which could change the electrical properties of the surfaces or those of the environment. It shall be sufficiently well resolved in space to separate the effects of the charging of various elements (radomes, fairings, antenna covers, canopies) on each navigation/communication subsystem, and to permit the observation of local sparks due to a faultly bonding.

The validity of the simulation (c) shall be confirmed by visual/auditory observation of the behavior of the navigation/communication instruments. Under such simulation tests and for a given electrostatic protection, an experienced pilot, seated in the cockpit, wearing earphones and watching the displays, shall identify the situation as similar to a real flight with different atmospheric conditions.

III. VERIFICATION OF BONDING

In the case of static electricity elimination, bonding requirements are by no means stringent : two conductors can be considered as bonded if the resistance of their connection is lower than $10^5\ \Omega$. This relatively high value permits a capacitive type of measurement, with injection of a measuring current through the thin insulating layer of paint which protects the aircraft skin. An instrument has been built at ONERA for this purpose ; its description is beyond the scope of this paper and only its performance will be mentioned. When the sensing head of the instrument, set on the "bonding mode" position, is pressed against the insulating layer of paint covering a given metallic panel, and a connection is made from this head to the structure of the aircraft, a meter gives the order of magnitude of the resistance between the metallic panel and the aircraft structure : smaller than $10^5\ \Omega$, between $10^5\ \Omega$ and $10^8\ \Omega$ or higher than $10^8\ \Omega$. This indicates a correct electrostatic bonding, a faultly bonding or a complete insulation. The same type of measurement can be performed in the case of resistive antistatic coatings covering an insulating substrate and covered by an insulating layer of protecting paint : from the measured value, the quality of the bonding can be evaluated if the surface resistance and the geometry are known.

The advantages of this method lie in the fact that the protective paint has not to be removed, and also in its easy application ; moving the sensing head of the instrument from place to place in the same way a medical doctor uses a stethoscope gives immediately the answer about the quality of the bonding of the different metallic /resistive· panels of the aircraft surface.

Figure 1 is a photograph of the sensing head of the prototype of this instrument.

IV. SURFACE RESISTANCE MEASUREMENTS

An extension of the same principle to surface resistance measurements has also been done at ONERA. In the case considered now, a resistive layer has been deposited over an insulating composite panel. A thin insulating layer of paint covers the resistive layer, for the sake of protection against erosion by atmospheric agents (and also for aesthetic reasons .....). What we want is to measure the surface resistance of the resistive layer without destroying the protective layer of insulating paint.

At this time, it is interesting to note by which mechanism a resistive layer can operate properly in draining the electric charges from a surface even in the case when it is covered by an insulating layer. As a matter of fact, the surface layer cannot be traversed by the accumulated charges. However, the resistive layer, which is connected to the metallic structure at its boundary, permits the circulation of electric charges of opposite sign. These image charges form with the accumulated charge a double layer with no external field. The only possibility of breakdown is then an extremely short spark across the protective paint

when a critical voltage is reached. As a consequence of this spark, the accumulated charges recombine with their images. The radiating dipole corresponding to this spark is very small compared to the radiating dipole of a streamer flashing across the whole surface. More work is needed to analyze under which conditions an insulating surface which has suffered a previous breakdown is modified by multiple punctures, each minute hole due to a previous spark acting as a transverse short circuit, thus reducing the surface resistance of the protective paint without changing its mechanical and optical properties.

Coming back to the surface resistance measurement, and recalling that its detailed description is beyond the scope of the present paper, it suffices to say that surface square mesh resistances ranging from $10^4\ \Omega$ to $10^{10}\ \Omega$ can be measured by this instrument. For this purpose, the sensing head previously mentioned is pressed against the radome, fairings, canopy or antenna cover to be tested, and the instrument is set on the "surface resistance mode" position. The value of the surface resistance is given directly by the meter (or can be computed from the value given by the meter if the geometry of the electrodes arrangement is taken into account). Depending on different versions of the instrument, a bonding connection to the aircraft metallic structure can be needed or not ; this connection is in any case needed for using the instrument in the "bonding" mode, which is generally required when a complete set of measurements is to be taken.

V. SIMULATION OF TRIBOELECTRIC CHARGING

If the aircraft to be tested were perfectly conducting all over its surface area, the charges induced by friction or collected in the atmosphere would be distributed according to the surface geometry. In this case, it would be sufficient, for performing a check on corona discharges location and on the associated R.F. noise, for example, to connect the surface to a high voltage generator, with the aircraft insulated. As field configuration can be roughly computed from model measurements, this kind of test is necessary only to verify that the passive dischargers are performing correctly, i.e. that all the corona discharges are located at their tip. We will come back to this type of measurement in the next section.

The real problem comes from the fact that no aircraft is perfectly conducting all over its surface, as radomes, canopies, windshields, fairings and antenna covers are insulating when not specifically treated to acquire total or partial surface conductivity. What we want to know are the following two answers.

a) For a given untreated dielectric surface, what are the consequences of a given local charging current on the operation of the navigation/communication subsystems of the aircraft ? This includes at once the anomalies induced, on the corresponding antenna, by charging its dielectric cover and by charging the other antenna covers.

b) If the same dielectric surface has been treated, is the applied treatment sufficient for complete elimination of all the observed anomalies, over the whole range of charging currents likely to be experienced in flight ?

By stressing that the source of radioelectric disturbances is essentially due to the local streamers associated with the redistribution of charges accumulated on the insulating surfaces [4, 5, 6], it is easy to see that local charging of these surfaces will produce approximately the same effect if the aircraft structure is grounded through a high resistance or if the aircraft is in flight. This remark opens the way to a very important test, i.e. the observation of the effects of local streamers on the performance of navigation/communication subsystems, and for a complete verification of streamers disappearance when the insulating surfaces have been duly treated. What one needs for performing this test is a good source for charging locally the insulating surfaces.

Taking into account the order of magnitude of the charging current density to be simulated, and also its very nature (surface charging by low energy and not by energetic particles), two types of processes can be used :

a) triboelectric charging, using a two-phase flow of air with a suspension of uncharged fine particles ; this method has yielded interesting results in the laboratory [6] [7] ; however the particles are not drained by the airflow as in flight, and their accumulation on the surfaces raises a serious problem ; that is why this method has not been used in this work ;

b) charge collection using an injector of charged particle ; this method is good as long as an efficient charge injector is used ; it is this method that has been applied at ONERA, as described below.

Instead of using very high voltage generators, ONERA has extended a method proposed first by Whewell, Bright and Makin [8] following a previous work by Marks, Baretto and Chu [9]. In the corresponding device, low mobility charged microparticles of water, obtained by condensation of humid air expanding in a supersonic nozzle, and driven by fluid friction in the jet, are used as charge carriers. They drain electric charges from an aircraft in spite of the strong associated space charge electric field. The same principle is applied here to inject charges upon an insulating surface, in spite of the associated repulsive electric field. Note that the microparticles of water evaporate after leaving the supersonic region of the jet ; this simulation method is therefore clean in that sense that the local properties (surface resistance and breakdown voltage) are not modified when the charge injector is operating.

The detailed description of the device, with the modifications and improvements brought by ONERA for increasing the injection current over the values obtained previously, will be given elsewhere.

Figure 2 is a schematic representation of this charging device. Figure 3 is a photograph of the instrument. Its main characteristics are listed in Table I. Figure 4 shows how this injection is performed when testing a real aircraft ; a first operator applies the charged flow upon a given antenna cover and, at the same time, a second operator watches the displays associated with the navigation equipment of the aircraft and estimates the noise collected by the communication receivers. Visual observation of induced sparking or streamers can also be performed (fig. 5). The setting of the injected current permits one to modify the charging current density, ranging from low to high for simulation of weather conditions between fair and extreme. The difference between the case of untreated and treated surfaces is striking and points out the importance of having a carefully applied surface treatment ; associated with the surface resistance and bonding measurements, this simumation permits also a quantitative

correlation of the reduction of the surface resistances (or the improvement of the bonding) with the decrease of the noise induced by aircraft charging on navigation/communication subsystems.

To apply this method one needs a supply of compressed air with a pressure of 6 bar, and of a power supply with a voltage of 10 kV, positive and negative. Both can be readily operated in the laboratory, in the factory and in the airfield. As shown in figure 4, the injection nozzle is of light weight, and can be handily used to charge any part of the aircraft. An important detail must be pointed out : in this experiment, the radioelectric disturbances are produced by local discharges following charge accumulation on insulating surfaces. Assessment of the noise produced by these discharges is possible only if the injector itself does not transmit any discharge noise; this property is an important adavantage of the device described, and its achievement was the result of a careful design involving a difficult optimization process.

VI. MEASUREMENT OF DISCHARGER/ANTENNA COUPLING

The last verification is related to the coupling between the corona discharges initiated at the tip of the passive dischargers and the various antennas of the aircraft. The first step of this measurement follows directly the method introduced by J. Nanevicz [1]. The aircraft being grounded through a high value resistor (to avoid RF current circulation in ground lines), a mobile corona generator or a calibrated RF transmitter is operated at various positions where coronas can be located in flight, and the RF noise induced on the aircraft antennas is measured. In this measurement, the d.c. current circulation is not relevant: that is why the aircraft is essentially at zero potential. The second step of this measurement consists on a verification of the noise induced by the dischargers fitted in the aircraft. This is readily done by blowing with the current injector directly over the discharger tip ; the conona is initiated by the space charge electric field, and its current can be set in changing this space charge, i.e. by varying the current of the injector. If necessary, a third step can be performed to ensure that no corona is generated in unwanted places. This step is more difficult, in that sense that it implies the insulation of the aircraft with thick dielectric slabs and application of a high voltage to the structure.

For all those tests, the coupling is measured in terms of induced noise for each frequency range of interest (with a fieldmeter or a spectrum analyzer), or in terms of errors brought to a given navigation/communication subsystem.

VII. APPLICATION TO A REAL AIRCRAFT : TESTING THE FALCON 10 AND THE ALPHA-JET FOR STATIC PROTECTION

In order to verify the proposed methodology, it has been decided by STTA (Service Technique des Télécommunications de l'Air) to specially equip a laboratory aircraft, under ONERA guidance and with the help of CEV (Centre d'Essais en Vol, Brétigny) and of AIA (Ateliers Industriels de l'Air). This aircraft will be very helpful for performing in-flight overall tests for assessing the validity of the proposed methods of protection against static charging. The equipment of this aircraft is in the process of being installed and the test will proceed immediately ; it is therefore too early for reporting about this experiment.

Fortunately, thanks to an extremely efficient cooperation with the Company "Avions Marcel Dassault-Bréguet Aviation" (AMD-BA), preliminary answers have been already obtained with the Falcon 10 business aircraft and the Alpha-jet trainer ; for this latter aircraft, during a four-day operation in April 1978, the following items have been performed :

a) search for and listing of all faultly bondings ;

b) measurement of surface resistances of various insulating elements ;

c) observation and measurement of radioelectric disturbances brought by aircraft charging to the VHF subsystems (VHF communications, VOR and ILS).

The methods for alleviating the noise induced by aircraft charging are being applied to a Falcon 10 laboratory aircraft ; evaluation of the residual noise will follow, with a view to assessing the possibility of fitting the "export" aircraft with an automatic direction finder. It is now well known that the automatic direction finder is the piece of navigation equipment most sensitive to radioelectric disturbances (at the same time, it is the cheapest and the most useful for flights over developing countries which are not equipped with more sophisticated navigation aids).

During this operation, the aircraft was in flight configuration, with engines off. The wheels were supported by insulating slabs, as was the generator supplying the 27 V d.c. voltage needed for operating the aircraft equipment. A 8 MΩ bleeder resistor was inserted between aircraft structure and ground.

The equipment used for the experiments was the following :

a) the surface resistance meter, operating in both "bonding" and "surface resistance" modes ;

b) the charge injector ;

c) a fieldmeter, giving the r.m.s. value of the radioelectric noise in selected bandwiths ;

d) a spectrum analyser.

Figure 6 is a general diagram of this experiment. The temperature during the tests was approximately 18°C and the relative humidity ranged between 50% and 65%.

The first operation consisted of searching for the fautly bondings ; it was achieved in a few hours for the entire aircraft. The elements with a faultly bonding were easily located and listed.

The second operation was the measurement of surface resistances of various elements. Some representative values are listed in Table II.

The third operation involved the use of the charge injector for studying the induced disturbances on navigation/communication subsystems. In a first step, the disturbances were qualitatively detected by auditory observation ; in a second step, the degree of interference was measured by the fieldmeter ; in a third step, some characteristic spectra were observed with the spectrum analyzer. During these measurements, the charge injector was operating at a distance of 150 mm from the impacted surfaces ; the charging current was of the order of 50 μA.

A number of observations were collected. The most sensitive zones were found, within a few hours, to be the untreated cover of the VOR-ILS antenna, the canopies, the dischargers with their decoupling resistor shorted, and, to a lesser extent, some critical zones where corona was initiated. It was observed that no disturbance was generated by charges impacting upon a properly treated antenna cover.

Table III gives the induced noise in a VHF antenna for various configurations. The results show that the charging of a given untreated antenna cover generates noise not only for the corresponding antenna but also for the antennas located at some distance. This table gives also the noise of a discharger with its decoupling resistor in good condition or shorted, and the noise generated by impacting the canopy close to the interface with the structure. Figure 7 is an example of noise spectrum variations obtained when the impacting current increases from 10 to 50 μA.

One of the most interesting results concerns the VOR-ILS operating with an untreated antenna cover. Three observations can be stated in this case :

a) for an input current larger than 20 μA, ILS signal identification becomes impossible ;

b) in the same condition, the information brought by the localizer and glide path receivers becomes wrong : for any initial position of the panel display, corresponding to a given ILS signal, increasing interference brings the display in the null position ; if the rate of charging of the aircraft is increased, the flag associated with the fault disappears : everything seems to indicate perfect landing conditions ;

c) when an antenna cover is treated, the above mentioned disturbances disappear : charging the treated cover of the VHF antenna does not induce any noise in the VHF subsystems.

The very dangerous situation reproduced in (b) is by no means specific of the Alpha-jet : it is likely to occur for any ILS system with an untreated antenna cover. This last example shows that, by using the proposed method of testing aircraft charging, dangerous anomalies in the behavior of navigation/communication subsystems operating in adverse weather conditions can be discovered and eliminated when the aircraft is in the laboratory, in the factory or on the field.

VIII. CONCLUSION

This paper has presented a procedure for testing the vulnerability of a protected or unprotected aircraft to static charging, and for assessing the validity of the methods used for this protection.

It is hoped that careful application of this test procedure shall, in the near future, increase the reliability of navigation/communication subsystems during operational use.

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Characteristics	
Generating pressure	$\sim$ 5 bar
Flow rate	$\sim$ 20 Nm³/h
Power supply voltage	$< \pm$ 10 kV
Nozzle diameter	2.3 mm
Current	0 à 50 μA
Two polarities	

Table I – Active discharger.

Table II – Surface resistances of various elements.

Treated VHF antenna cover on fin top	$10^6 \leq R_0 \leq 10^7$ Ω
Protective paint :	$R_0 \simeq 2.10^8$ Ω
· wing panel and upper fuselage (green paint)	$R_0 \simeq 10^{11}$ Ω
· lower fuselage (grey paint)	$R_0 \simeq 2$ à 5.10^{10} Ω

Table III – Charging current and noise level in a VHF antenna for various configurations.

Experimental configuration	Charging current	Noise level
1. Charge applied to the untreated cover of the antenna	0 μA 10 20 30 40	2,5 μV 22 50 141 158
2. Charge applied to the treated cover of the antenna. Graphite paint visible (no protective paint)	from 0 to 80 μA	0,5 μV
3. Charge applied to the untreated cover of the VOR antenna. The sensing VHF antenna has a treated cover	0 μA 10 20 30 40	2,2 μV 12,5 63 200 355
4. Charge applied to a discharger close to the cover of the VHF antenna - discharger in good condition - discharger with its resistor shorter	 50 μA 50	 2 à 4 μV $\simeq$ 1000 μV
5. Canopy charging	50 μA	320 μV

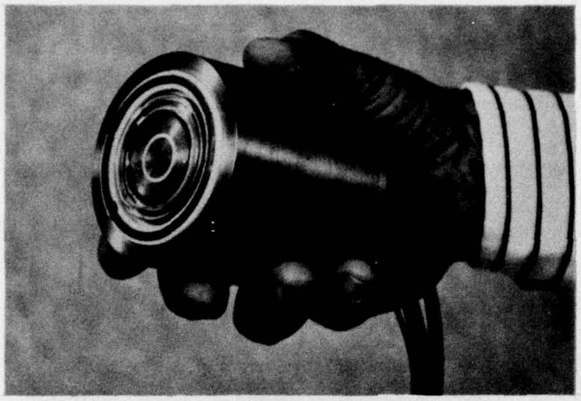


Fig. 1 – Sensing head of the surface resistance meter.

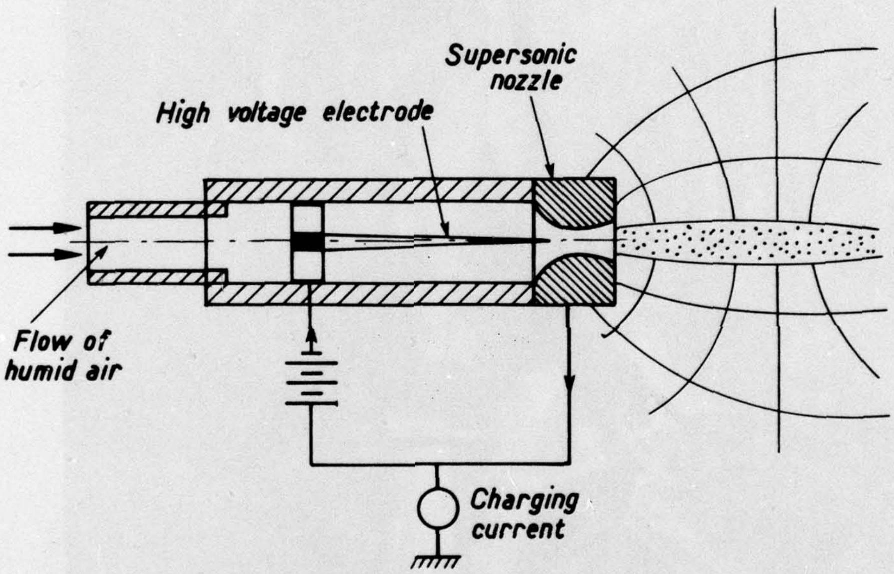
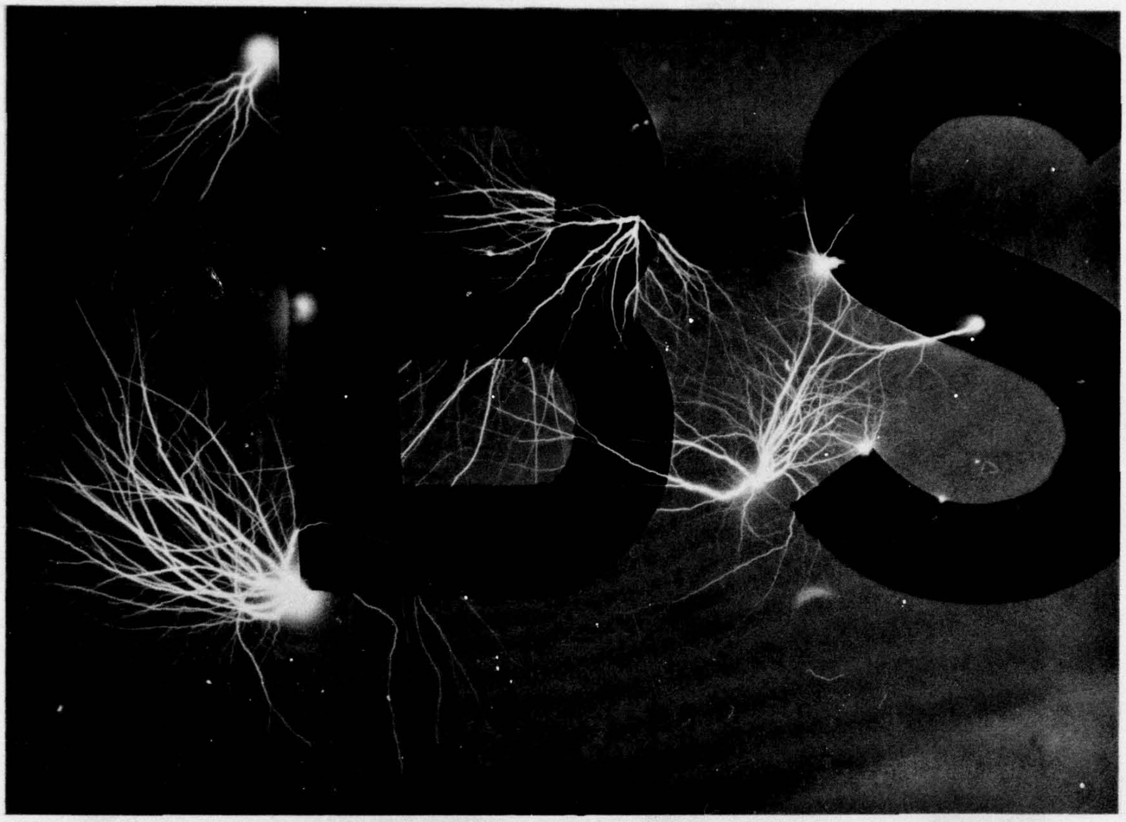


Fig. 2 – Diagram of the charge injector.

Fig. 3 – Charge injector with its power supply.

(Photo AMD-BA)

Fig. 4 – Simulating static charging with an aircraft on the ground.



(Photo AMD-BA)

Fig. 5 — Surface streamer produced by charging the fuselage.

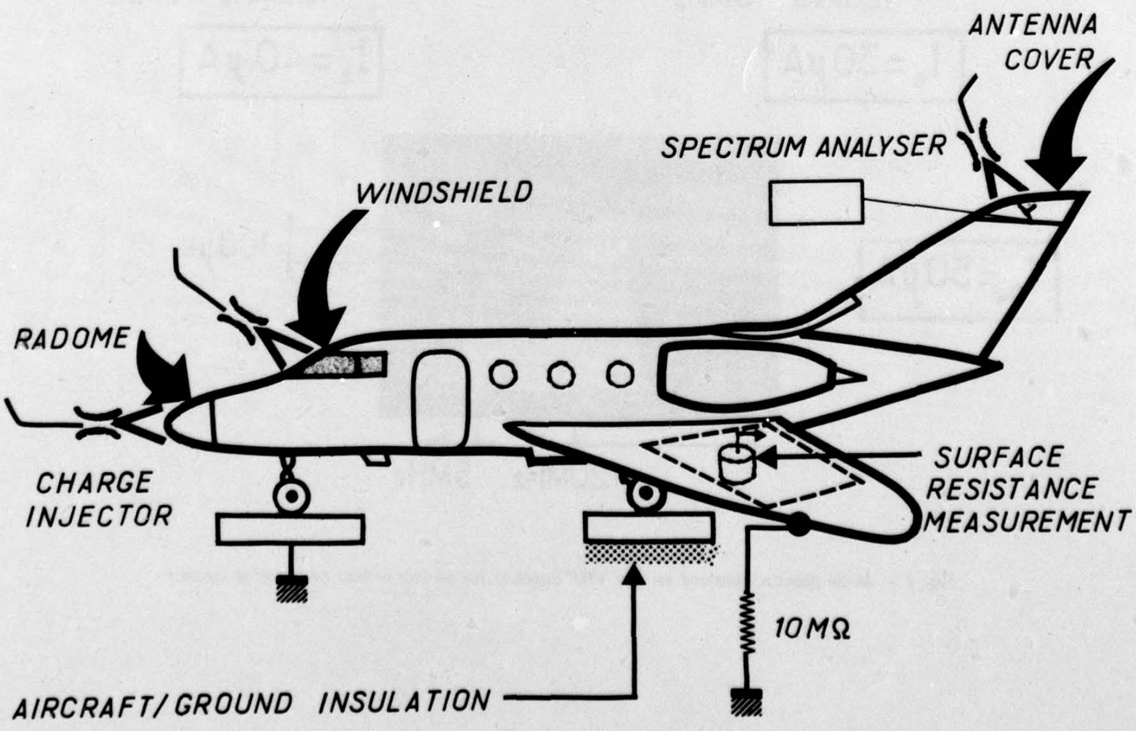


Fig. 6 — Block diagram of the experimental set-up used for aircraft testing.

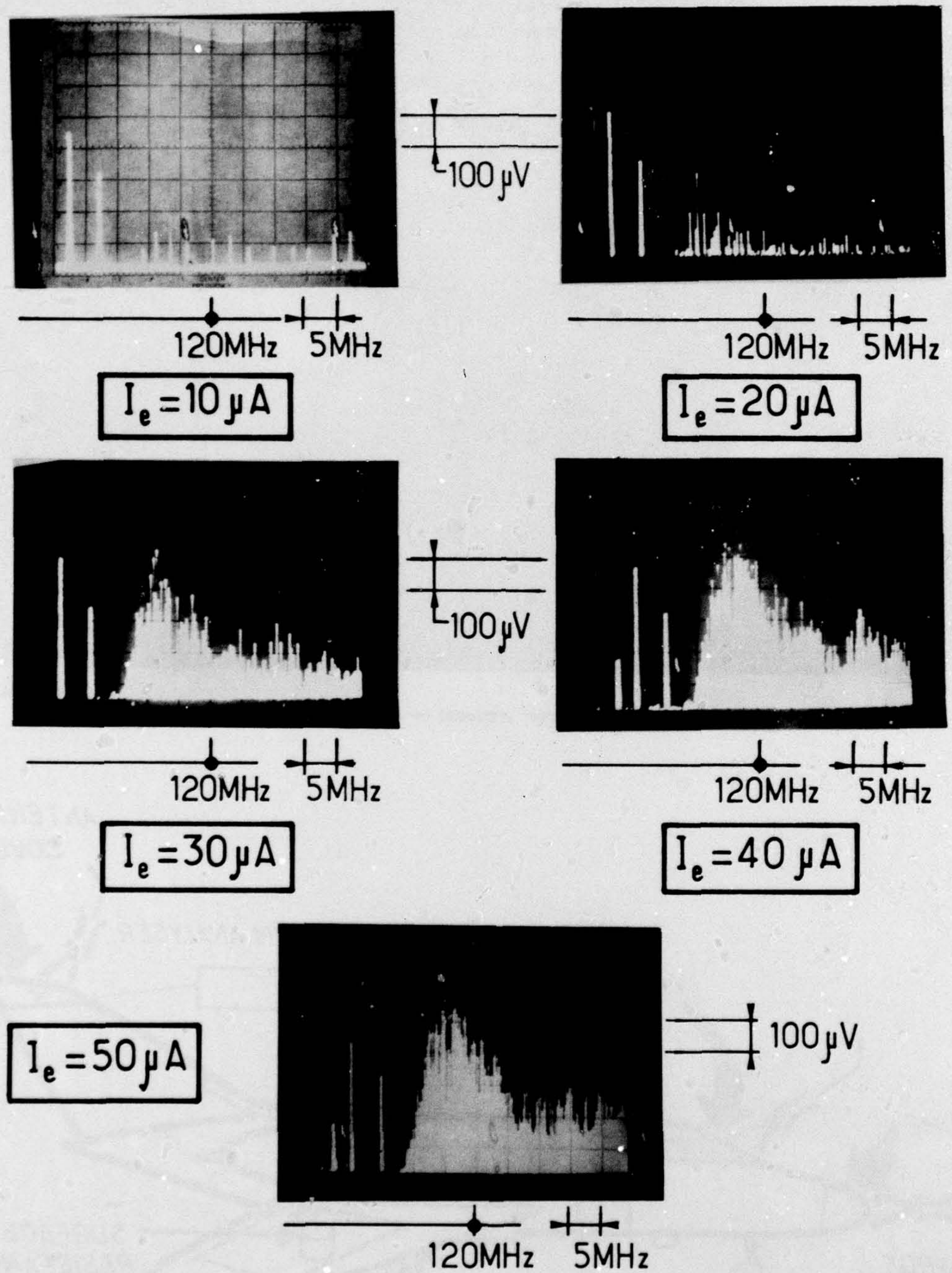


Fig. 7 – Noise spectra obtained on the VHF antenna for various values of charging current.

BIOGRAPHICAL NOTE

J. TAILLET

Joseph Taillet received his engineering degree from Ecole Supérieure d'Electricité in 1947 and his Science Doctorate in Physics from Paris University in 1964. He served as research engineer with the Commissariat à l'Energie Atomique, where he was engaged from 1948 to 1958 in the construction of the first generation of high voltage and high energy particle accelerator built in France, and where he was in charge, from 1959, of the Plasma Physics Division in Saclay. In 1965-1966, he visited the Electrical Engineering Department of MIT as Associate Professor. He is since with ONERA, as Scientific Director of the Physics Department. Dr Taillet is a member of the Avionics Panel of AGARD, of the Plasma Physics Commission of the International Union of Pure and Applied Physics, and of the Scientific Programme Committee of the Centre National d'Etudes Spatiales (French Space Agency).

CLOSING REMARKS

by J.H. Belanger (CA)

Move over Jupiter! That could be a fitting motto for these valiant researchers in lightning and static who have laboured for many years in numerous places to observe lightning, to detail its processes, to generate it under control to aim it at a target, to study and discuss its effects, and to provide advice for protection against it, notably to designers, maintainers, and operators of aerospace vehicles.

This conference at the Office National d'Etudes et de Recherches Aérospatiales on Certification of Aircraft for Lightning and Atmospheric Electricity, following in the tradition of previous conferences on Lightning and Static, at the Culham Laboratory in 1975 and before that in the United States, has provided an excellent forum where experts in this field could share their views, and present data that could serve as a basis for the evaluation of sound standards concerning structures and testing, that can add to the security of air travel and of other aerospace operations.

As a long-time participant for Canada in the MAS/AEWP (Air Electrical Working Party) a NATO body which is engaged in developing international Standardization Agreements, I want to thank those who have prepared papers for presentation at this conference, which dealt with the various manifestations of lightning and its effects, direct and indirect, as it may affect the Aerospace industry, knowing that your present efforts continually advance the state of the art.

On behalf also of all national delegates to the AEWP, which is soon to hold its 14th Meeting at l'Etat - Major de l'Armée de l'Air, I do express our appreciation to our hosts at O.N.E.R.A. and to the chief conference organizers, Mr. Gary DuBro and Mr. Charles Seth of the USAF Aeronautical Systems Division, and in particular to Dr. Joseph Taillet, Scientific Director of Physics at ONERA for his spirited role in holding these proceedings at this Aerospace Research Facility of the French Government.